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Technical Report 15

HELICOPTER STRUCTURES – A REVIEW OF LOADS, FATIGUE DESIGN TECHNIQUES AND USAGE MONITORING

by

D.C. LOMBARDO

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SUMMARY

This report is a review of traditional practice with respect to helicopter structural integrity. Aspects covered are: significant fatigue loads for the airframe and rotor system, the methods used in the fatigue design of current and previous generation rotorcraft, fatigue test requirements, and health and usage monitoring methods.

DSTO
AUSTRALIA

POSTAL ADDRESS: Director, Aeronautical Research Laboratory,
506 Lorimer Street, Fishermens Bend, 3207
Victoria, Australia.

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1. INTRODUCTION

The Aeronautical Research Laboratory (ARL) of Australia's Defence Science and Technology Organisation (DSTO), has a long history of providing structural integrity advice and support to the various services of the Australian Defence Force (ADF), in particular the Royal Australian Air Force. This support has been mainly concentrated on structural integrity aspects of the fixed-wing aircraft operated by the ADF. Requests for support for the structural integrity of ADF helicopters have beetn minimal*. This situation is set to change since the ADF has recently procured a fleet of 39 Sikorsky S-70A-9 Black Hawk and 16 Sikorsky S-70B-2 Seahawk helicopters. The total cost of this procurement is over $1000m and hence it represents a significant investment for both the ADF and Australia. To achieve cost-effective and safe operation of these helicopters during their operational life, the ADF will need the type of back-up that ARL has consistently provided for fixed-wing aircraft.

To provide this support, DSTO needs to acquire in-house knowledge and expertise concerning the structural integrity aspects of helicopters. This report is one of the first steps in acquiring such knowledge and, as such, it is a review of the traditional practice in the helicopter world as applied to metallic structures. Neither the use of advanced composite materials in helicopters nor the use of damage tolerance techniques are addressed in this report.

The scope of this report can best be summed up in the questions that prompted it:

- What are the fatigue damaging loads experienced by helicopters and in what ways are they different from those experienced by fixed-wing aircraft?

- How have helicopters traditionally been designed for fatigue?

- What components are particularly prone to fatigue failure?

- What has been done in the field of helicopter monitoring?

This report has been written so as to be readily understood by those with a knowledge of both aircraft and aircraft fatigue. No specialised knowledge of helicopter terminology is required as a glossary at the end of this report provides definitions for some of the more common terms peculiar to helicopters.

* ARL has provided the ADF with help in other areas of helicopter operations such as helicopter flight behaviour and performance, the maintenance and care of engines and transmissions, human factor elements such as cockpit ergonomics and crew workload, and helicopter accident investigations.

2. BACKGROUND

There are many different configurations possible for helicopters, or rotary-wing aircraft as they are also known, (see Fig. 1, next page), but by far the most common is the type that uses a single main rotor for lift and propulsion and a tail rotor to provide an anti-torque force (Fig. 1(a)). The term "rotary-wing aircraft" implies that such aircraft have rotating wings to generate lift, unlike other aircraft that have wings rigidly fixed to their fuselages. Individually known as rotor blades, collectively as a rotor, it is these rotating wings that give helicopters both their versatility and a susceptibility to fatigue problems that is very different to that found in the world of fixed-wing aircraft.

The question might well be asked, "Both helicopters and fixed-wing aircraft are heavier-than-air flying machines, with all which that entails, so why is helicopter fatigue and structural integrity so different from the fatigue of fixed-wing aircraft?". The answer lies in the fact that the generation of lift via rotors rather than fixed wings produces a loading environment that is ruled by large dynamic loads that are applied at a high rate. In fact, it has been suggested, perhaps unkindly, but with some truth, that the easiest way to perform fatigue tests on helicopter components is to install them on a helicopter and let it apply the fatigue loads.

At the heart of a helicopter, both physically and in terms of fatigue problems, lie the engines, transmissions, drivetrain, and rotors. These, in themselves, consist of a multitude of components that are highly loaded both in the magnitudes of the applied loads and the number of applied cycles. The failure of any one of these components can be, and often is, catastrophic, as distinct from fixed-wing aircraft where the number of critical components is less and structural redundancy is easier to build in. A survey of serious accidents involving fixed and rotary wing aircraft[1,2] that occurred in the period 1927 to 1981 lists a total of 1466 accidents for fixed-wing aircraft and 419 for rotary-wing. If the data prior to 1964, as well as all military data, are ignored* then the numbers become 1191 and 354 respectively. To put this into perspective, it is useful to compare the number of accidents, and resulting fatalities, against the number of aircraft in the world during those years (see Table 1, page 4). As can be seen, helicopters suffer a proportionately greater number of accidents and fatalities than fixed-wing aircraft. Torkington[3] highlights this same difference regarding civil aircraft in Australia. There are, of course, other factors apart from the number of aircraft to be taken into account when comparing accident data such as the number of flights or hours flown, or the types of missions performed. However, even when such factors are included, helicopters still come out worse[4].

The breakdown of fatigue-related accidents listed in the above mentioned survey is: structural 57% (rotors 37%, fuselage 8%, landing gear 3%, flight controls 9%), engines and transmissions 32%, and other causes 11%. In terms of fatigue therefore, rotors, engines, and transmissions are the areas which experience the majority of fatigue failures in helicopters and consequently most of the research is concentrated in these three areas.

Throughout this report, several helicopter types are used to illustrate various points. For simplicity, they are mostly referred to by their names or type designations (e.g. AH-64, Sea King, Black Hawk). Table 2 contains an explanation of the names and type numbers used in this report. For further information the reader is referred to books which deal specifically with helicopters and their history[5,6,7,8].

* In the survey, it is stated that the data prior to the years around the mid-sixties are far from comprehensive. As well, only a few military services responded to the requests of the survey's authors for information.

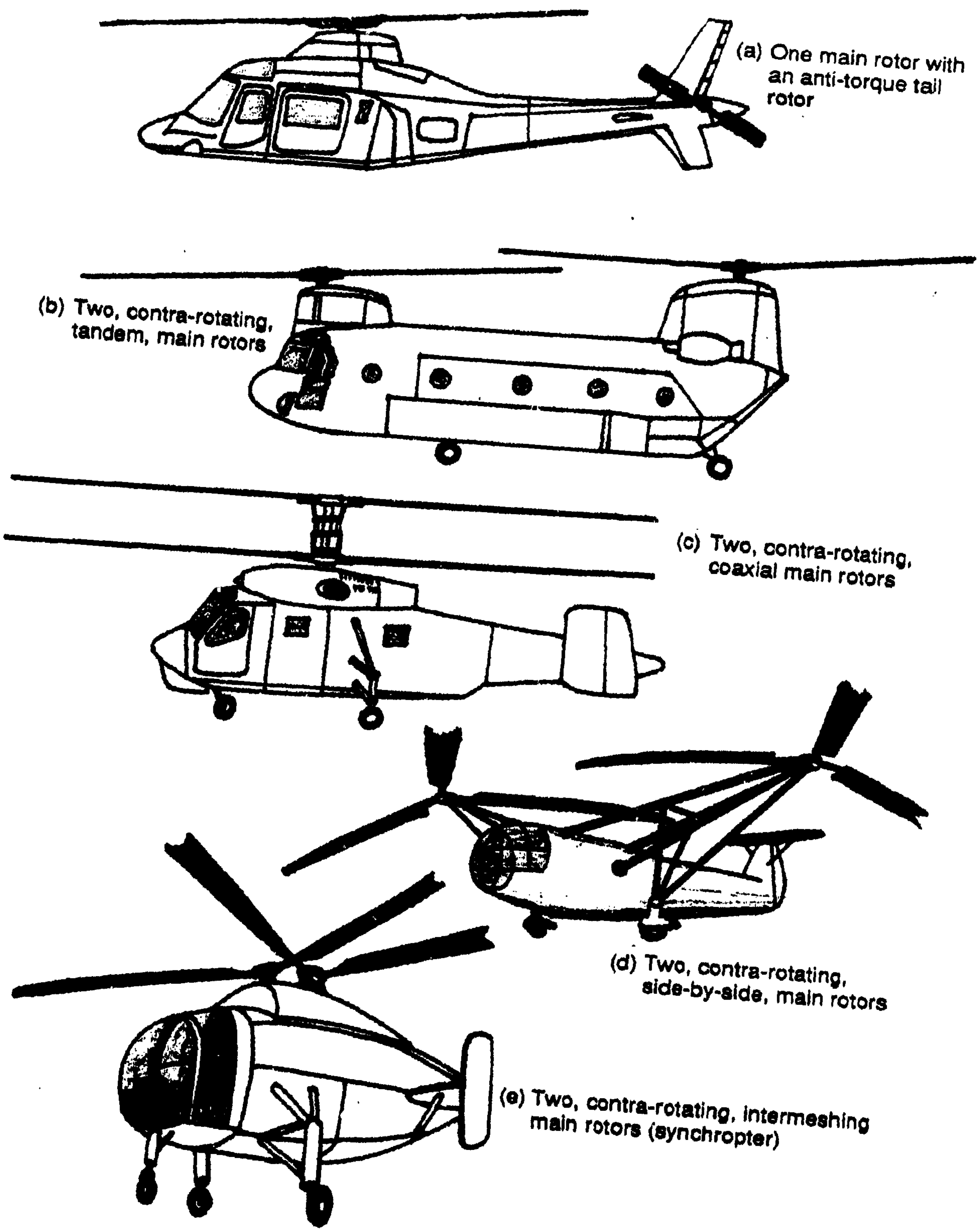

Figure 1: Some helicopter configurations

Table 1
Accident Rates for Fixed-wing and Rotary-wing Aircraft

Fixed-wing Aircraft

Year	1966	1967	1968	1969	1970	1971	1972	1973
Number of Aircraft	142480	156110	168320	179140	183740	215850	224600	238100
Number of Accidents	78	72	44	43	56	62	47	66
Number of Deaths	19	32	110	15	21	20	16	15
Accident Rate[1]	*0.055*	*0.046*	*0.026*	*0.024*	*0.031*	*0.029*	*0.021*	*0.028*
Death Rate[2]	*0.013*	*0.020*	*0.065*	*0.008*	*0.011*	*0.009*	*0.007*	*0.006*

Year	1974	1975	1976	1977	1978	1979	1980	1981
Number of Aircraft	250550	264840	279770	293120	309200	323570	337180	343170
Number of Accidents	66	68	71	61	71	62	68	64
Number of Deaths	29	38	158	16	21	36	104	12
Accident Rate	*0.026*	*0.026*	*0.025*	*0.021*	*0.023*	*0.019*	*0.020*	*0.019*
Death Rate	*0.012*	*0.014*	*0.056*	*0.005*	*0.007*	*0.011*	*0.031*	*0.003*

Rotary-wing Aircraft

Year	1966	1967	1968	1969	1970	1971	1972	1973
Number of Aircraft	2950	3660	4060	4550	4780	6550	7220	8050
Number of Accidents	24	16	16	12	10	8	19	19
Number of Deaths	22	12	26	8	2	4	60	18
Accident Rate	*0.814*	*0.437*	*0.394*	*0.264*	*0.209*	*0.122*	*0.263*	*0.236*
Death Rate	*0.746*	*0.328*	*0.640*	*0.176*	*0.042*	*0.061*	*0.831*	*0.223*

Year	1974	1975	1976	1977	1978	1979	1980	1981
Number of Aircraft	9270	10150	11060	11940	12750	13750	15100	16580
Number of Accidents	15	27	22	32	33	25	23	28
Number of Deaths	19	17	21	12	30	19	22	9
Accident Rate	*0.162*	*0.266*	*0.199*	*0.268*	*0.259*	*0.182*	*0.152*	*0.169*
Death Rate	*0.204*	*0.167*	*0.190*	*0.101*	*0.235*	*0.138*	*0.147*	*0.054*

Notes:
1. Accident Rate is the number of accidents which occurred per hundred aircraft
2. Death Rate is the number of deaths which occurred per hundred aircraft

Sources:
A. Aircraft Numbers - Civil Aviation Statistics of the World, ICAO (International Civil Aviation Organization) Statistical Yearbook, Volumes for the years 1975 and 1982
B. Aircraft Accident and Death Statistics - Compiled from Refs 1 and 2

Table 2
Listing of helicopter names and type numbers as used in this report

Name	Type Numbers	Manufacturer	Helicopter Type
–	S-69 ABC	Sikorsky Aircraft, U.S.A.	Experimental
Apache	AH-64	McDonnell Douglas Helicopter Company (Previously Hughes Helicopters), U.S.A.	Attack/Anti-armour
Bell 47	–	Bell Helicopter Textron, U.S.A.	Light observation/liaison
Black Hawk	H-60, UH-60, or S-70A	Sikorsky Aircraft, U.S.A.	Tactical Transport
–	BO 105 or MBB 105	Messerschmitt-Bolkow-Blohm Germany	General purpose
Cheyenne	AH-56	Lockheed Aircraft Corporation U.S.A.	Attack/Anti-armour/Escort (Did not enter production)
Chinook	CH-47	Boeing-Vertol, U.S.A.	Tandem rotor, Assault/transport
Cobra	AH-1	Bell Helicopter Textron, U.S.A.	Attack/Anti-armour
Hook	Mi-6	Mil, U.S.S.R.	Heavy assault/transport
Lockheed 286	–	Lockheed Aircraft Corporation U.S.A.	Prototype, civil transport
Lynx	–	Westland Helicopters, U.K.	Multi-purpose
Seahawk	SH-60B	Sikorsky Aircraft, U.S.A.	General purpose maritime/Anti-submarine
Sea King	–	Westland Helicopters, U.K.	General purpose maritime/Anti-submarine (U.K. version of the SH-3)
–	SH-3	Sikorsky Aircraft, U.S.A.	General purpose maritime/Anti-submarine
Sea Knight	CH-46	Boeing-Vertol, U.S.A.	Tandem rotor, Assault/transport
Seasprite	SH-2F	Kaman Aerospace Corporation U.S.A.	General purpose maritime/Anti-submarine
Sea Stallion	CH-53	Sikorsky Aircraft, U.S.A.	Tactical transport
Squirrel	AS350	Aerospatiale, France (Now part of Eurocopter)	Light transport

3. HELICOPTER LOADS

The loading environment to which helicopters are subjected differs markedly from that of fixed-wing aircraft. Hence, before proceeding with a discussion of the various loading actions on a helicopter, the two most important differences between helicopters and fixed-wing aircraft should be stated.

- Considering the case of steady flight (a condition in which nearly all aircraft spend most of their time), the loads on a fixed-wing aircraft are essentially static in nature. On the other hand, helicopters experience a complex loading that is both dynamic and highly vibratory[9,10,11,12,13]. This is the case in all flight conditions for a helicopter other than vertical flight in still air, where it applies to a lesser extent.
- Unlike fixed-wing aircraft, the time spent in a flight condition is as important as the condition itself[14,15]. In other words, "*... whereas on a fixed wing aircraft one manoeuvre goes with one incremental load, a manoeuvre of a helicopter results in a number of incremental load cycles ...*" (de Jonge[9]). This time dependence is the result of the cyclical loading produced by a rotor blade as it rotates during flight.

3.1 HELICOPTER SYSTEMS

In discussions on helicopter loading, it is generally convenient to divide the helicopter up into its major systems. Following the example of Fraser[16], these systems are:

1. Engines
2. Mechanical drive system (gearbox, transmission shafts etc)
3. Rotor system (main and tail)
4. Flight control system
5. Airframe (fuselage, tailboom, undercarriage, auxiliary aerodynamic surfaces)

Items 1 and 2 are not within the scope of this report (which is concerned with structural integrity) and so will not be discussed here. Item 4, the flight control system is here considered to consist of all the non-rotating mechanical links between the pilot controls and the control surfaces, both rotating (rotor blades) and non-rotating (fins). As the elements of the control system transmit rather than generate loads they will not be considered in this section. The remaining two items will now be considered.

3.2 ROTOR SYSTEM

The rotor system is what gives helicopters their versatility, that is, their ability to take off and land vertically, to hover, and to fly sideways or rearwards. This versatility though, comes at a price, that being that the rotor system is an "*... extremely complex dynamic system of rotating blades with fundamental and higher order vibratory response, complex unsteady aerodynamics verging upon aeroelastic instabilities and blade stall phenomena ...*"[12].

Consider also the following two quotations:

(a) "*Unlike the relatively well established science of load prediction for fixed wing aircraft, the helicopter load prediction techniques are still in their infancy ...* [the designer] *simply must determine, or at least confirm, all loads by experimental flight loads surveys.*", Crichlow et al[12], (1967)

(b) "*... rotor blade loadings are difficult to accurately predict: all current airworthiness requirements specify that fatigue analyses must be based on measured loads.*", de Jonge[9], (1986).

Although separated by almost twenty years, the complexity of the problem has evidently allowed little progress to be made in rotor blade load predictions. This is confirmed by a recent study[17] in which four "state-of-the-art" rotor load prediction programs were compared with each other and against flight data. The results are not very encouraging to those used to similar fixed-wing studies.

The rotor system has been defined[18] as including the rotor blades, hubs, hub-to-blade attachments, dampers, rotating control system elements and any rotating anti-vibration features. As already explained in the Background section of this report, the rotor system can consist of many different configurations: a main and tail rotor combination or a twin main rotor combination for example. Although the discussion that follows refers specifically to helicopters of the main and tail rotor type, it is applicable to the other configurations as well.

3.3 ROTOR SYSTEM LOADS

The rotor system loads for the main or tail rotor are generated by the rotor blades. These loads consist of aerodynamic, centrifugal, inertial, and gravitational forces (i.e. the weight of the blades). The total load is made up of the sum of these forces and their respective proportions will vary under different flight conditions. For example, when the helicopter is on the ground with its rotor spinning but providing no lift, the predominant loads are centrifugal. If the helicopter is stationary on the ground with its engine stopped then blade weight is the only load. In this case, the weight produces loads which are reversed in sign when compared to normal rotor loads. An important difference between aircraft wings and helicopter rotor blades is that most of the lift force on a rotor blade is produced over the outer portions of the blade (Fig. 2).

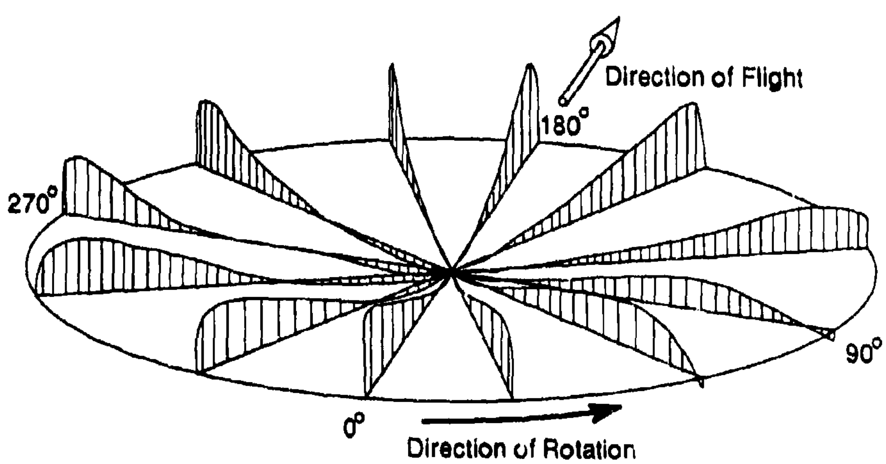

Figure 2: The distribution of lift along a rotor blade. Azimuthal angles are measured in the direction of rotation of the rotor blade with zero degrees pointing aft (from Ref. 9).

The centrifugal loads are important in determining the static loads in a rotor blade and are responsible for another key difference between helicopters and fixed-wing aircraft. On a fixed-wing aircraft in flight, the wing is placed in tension on one surface and compression on the other surface depending on whether it is in a positive or negative "g" manoeuvre. In contrast, a rotor blade has both its upper and lower surfaces in tension because the centrifugal (tensile) loads exceed the compressive blade bending loads.

In vertical flight, (including hover), both de Jonge[9] and Fraser[16] state that the loading is essentially static. Strictly, this is true only for a perfectly balanced, isolated rotor in a no-wind condition. For a real helicopter, in ambient wind conditions, the aerodynamic

interaction between the main rotor and the fuselage[19,20,21], and the interaction between the main and tail rotors, will produce unsteady loads in the rotor blades. As well, most tail rotors are mounted such that they produce rolling moments which must be counteracted by the main rotor thrust leading again to unsteady main rotor blade loads For helicopters with two main rotors, account must be taken of the interaction between the two rotors. The importance of these loads depends on the geometry of the helicopter fuselage, the spacing between the fuselage, the main rotor and tail rotor, and the strength of the ambient wind. In some cases, these unsteady loads can be significant in terms of fatigue damage.

In horizontal flight, each rotor blade now produces definite cyclic loads. That is, as each blade rotates about the main rotor shaft, the loads it generates vary markedly Figure 3 shows this variation in lift generated by a blade as it moves through a 360° sv). Figure 4 shows the breakdown of the lift into its harmonic components. Wha. his figure indicates is that not only do significant load variations occur at the rotor frequency, but also at its higher order harmonic frequencies[32,72]. This is confirmed by de Jonge[9] and also by Greaves[22] who states: *"The effect of these cyclic forces ... is to excite various blade vibrational modes resulting in significant oscillatory moments and torques on the* [rotor] *hub at frequencies other than* [the rotor frequency] *and with phase differences between each"*. This cyclic loading at and above the rotor frequency can produce a large number of loading cycles on the rotor system components. For example, consider a main rotor pitch-link attached to a rotor turning at 5 Hz* and assume that significant load variations are occurring at the rotor frequency. Thus, this pitch-link will experience 18000 load cycles during every hour of flight. Over the projected life of a helicopter (4500 hours for an Apache, 10,000 hours for a Seahawk or 20,000 hours for an SH-2F Seasprite[23], for example) such high frequency loading can result in around 10^7 to 10^8 cycles being applied to rotor system components. McDermott[24] describes a fatigue test on a McDonnell Douglas AH-64 Apache rotor blade in which he states that a damaged blade was subjected to over five equivalent flight hours of fatigue loading without failing. The actual number of cycles applied to reach the five hours was approximately 90000. This is quite different to the case of fatigue testing of fixed-wing aircraft where only a handful of cycles need be used to represent an hour of flight. In fatigue terminology, this type of loading is known as High Cycle Fatigue (HCF).

A helicopter designer would try to ensure that most of these cycles cause as little fatigue damage as possible. For steels, this would mean striving to keep stresses below their fatigue limits. However, the occurrence of large magnitude Low Cycle Fatigue (LCF) loads can cause the HCF loads to become more damaging than predicted. Moreover, unforeseen applications of LCF loads in conjunction with material and/or production defects can allow HCF loads to cause fatigue failures much earlier than expected[13,25]. LCF loads are generated by conditions such as autorotation and are discussed in the next section. Many components in a helicopter are subjected to both high and low frequency loads. For example, a main rotor blade sees both high frequency loads due to the variation in the blade lift and low frequency loads (e.g. the "on-off" centrifugal load). Other components, particularly in the fuselage may be subject mainly to low frequency loads in the same manner as a fuselage in a fixed-wing aircraft[26].

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* Main rotors typically rotate at 3 - 6 Hz (180 - 360 RPM), while tail rotors rotate at 15 - 30 Hz (900 - 1800 RPM)

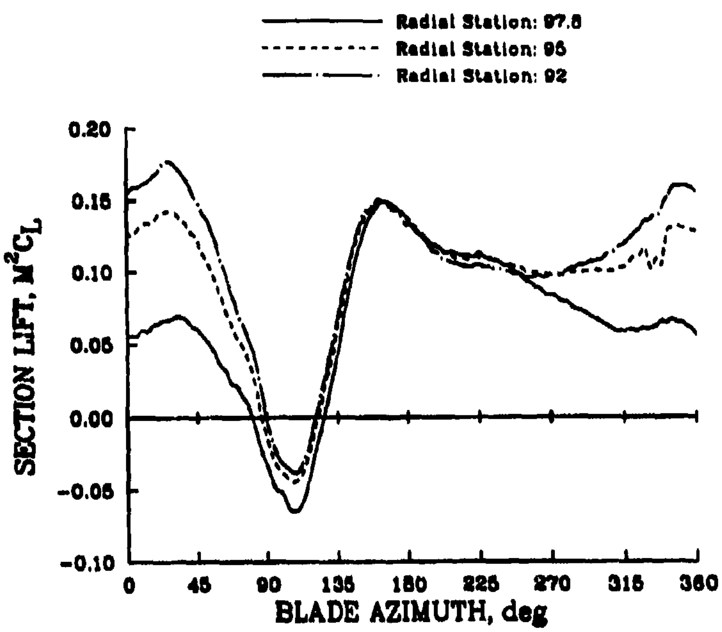

Figure 3: Variation in the lift produced at points (radial stations) along a rotor blade as it sweeps through 360°. The curves have been produced from actual flight data from a European helicopter. Radial stations are given in terms of percentage of blade radius. Azimuthal angles are measured in the direction of rotation of the rotor blade with zero degrees pointing aft.

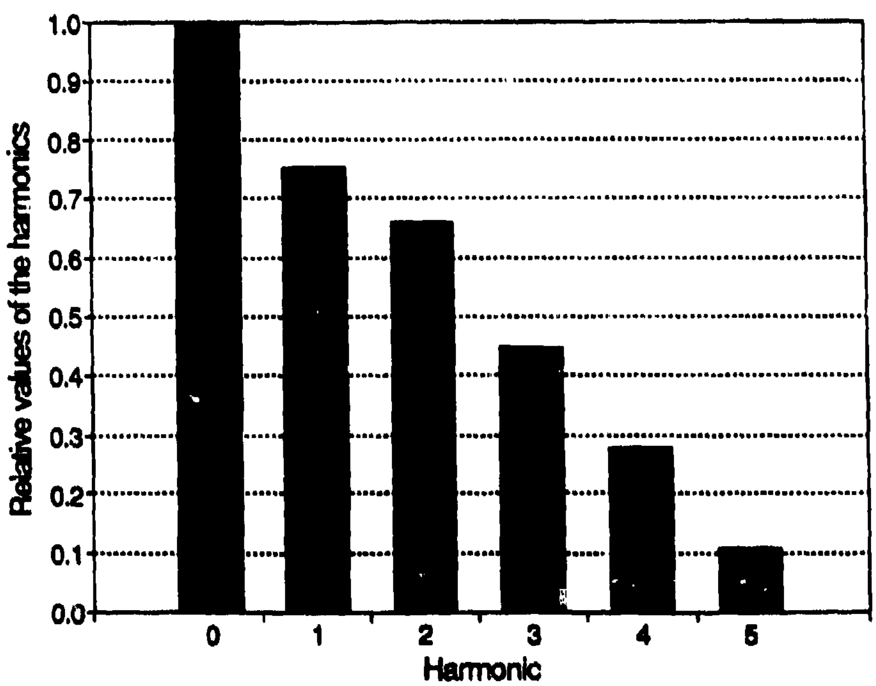

Figure 4: Comparison of the first to fifth harmonic components of the lift at station 97.8 (see Fig. 3 above) relative to the mean value (harmonic 0).

3.4 SIGNIFICANT MAIN ROTOR LOADS

There are several conditions and load cases which cause significant fatigue loading of the rotor system components and which allow normally insignificant High Cycle Fatigue loads to become damaging.

3.4.1 High Speed Flight

The severity of main rotor fatigue damage can be dependent on the helicopter speed as high speed flight is generally damaging. This is due to several factors:

(a) Aerodynamic Drag: The faster an aircraft flies, the higher its drag becomes, which means that more thrust is needed to propel the aircraft. A helicopter produces propulsive thrust by tilting its main rotor so that a component of its thrust is pointing in the direction of the desired motion. Therefore, as helicopter speed increases, so must the main rotor thrust in order to provide a larger horizontal force while still providing enough lift to overcome the helicopter weight. Therefore, as speeds rise, so too do the amplitudes and means of the cyclic loads applied to the rotor system components (Fig. 5). As indicated in the figure, at sufficiently high speeds, the lift at the tips of the blades can become negative.

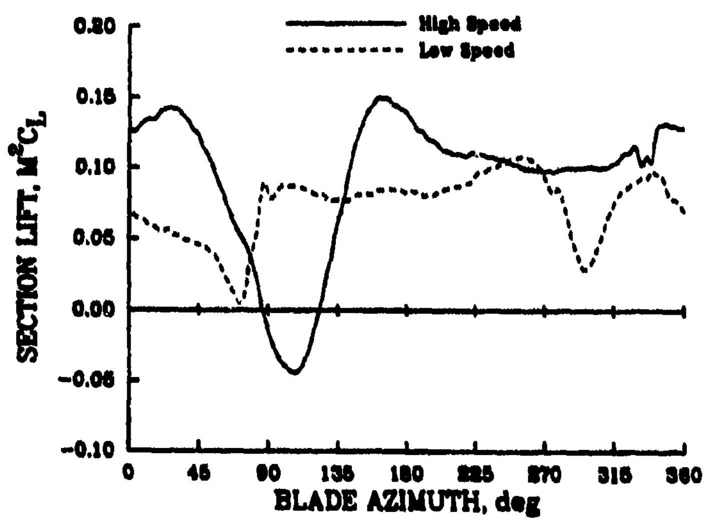

Figure 5: Comparison of the lift produced by a rotor blade for two different helicopter flight speeds. Lift is shown only at the 95% blade radial station.

(b) Compressibility Effects: At high forward velocities, the tip region of the advancing blade on a helicopter rotor can experience compressibility effects. Specifically, a large increase in drag is experienced as the tip speed approaches the speed of sound and consequently the blade aerofoil enters its drag divergence region. Figure 6 shows the region of the blade disc over which the large drag increases occur. Since this region exists only over a portion of the disc then highly vibratory loading is produced which has to be reacted by the rotor blade attachments at the blade root. Also, any shock waves generated in the tip region will also lead to vibratory forces which must also be reacted by the blade root.

(c) Stalled Flow: In forward flight, part of the retreating blade is stalled. At low speeds (Fig. 7(a)), this region is confined to the blade root (where it is due to reversed flow) and has a negligible effect on overall blade lift capability. As speeds rise, however, the stalled area increases in size and, as well, a stalled region begins growing at the tip (Fig. 7(b)). The effect of this is to produce large load cycles at the tip where the blade goes from high lift to stalled conditions during every revolution. Once again, vibratory loading is produced.

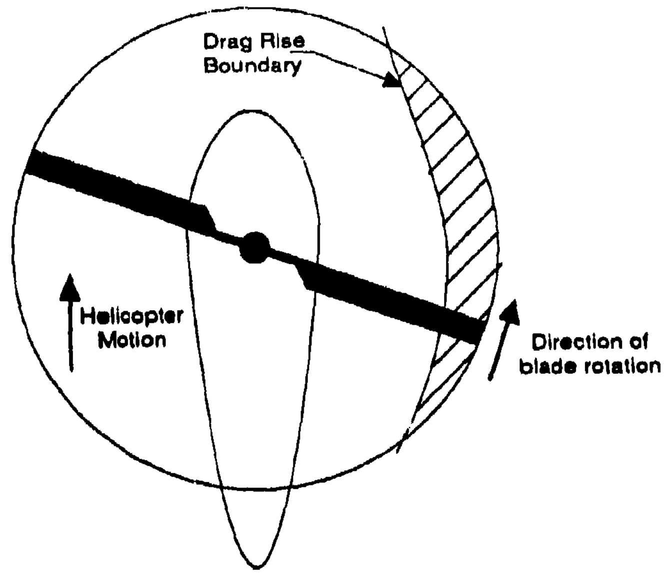

Figure 6: Typical extent of the rotor disc affected by compressibility effects during high speed flight.

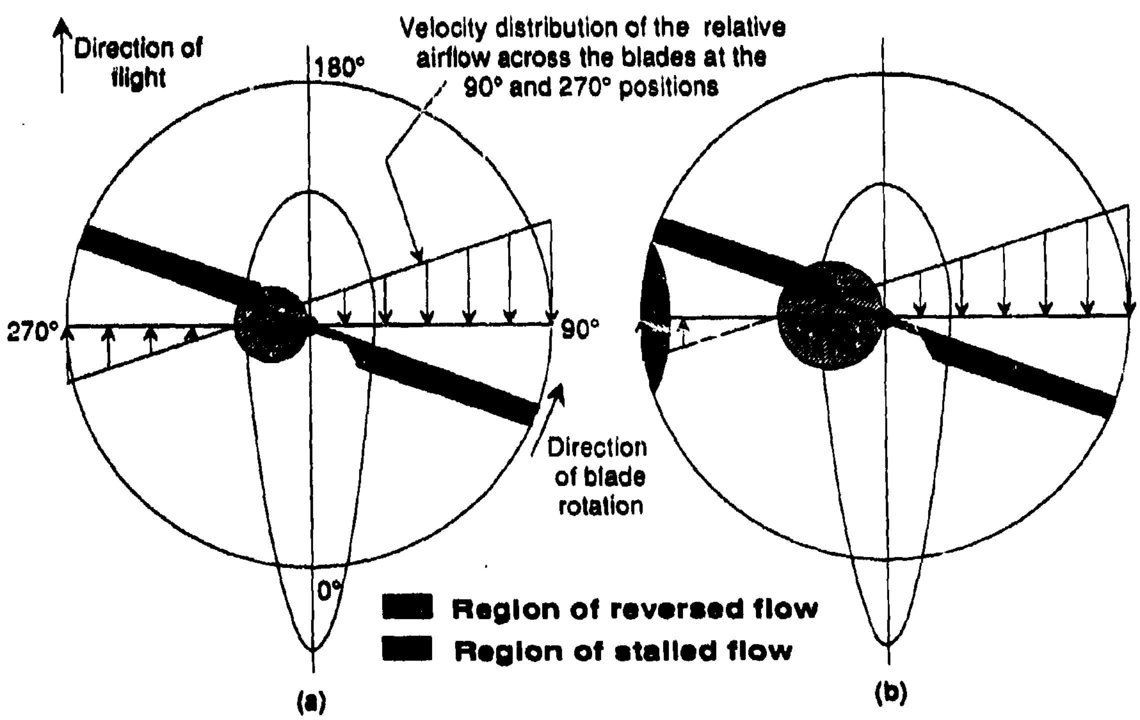

Figure 7: Regions of stalled and reversed flow through a rotor disc under (a) low speed and (b) high speed flight

Flight test results[12] for a Lockheed Model 286 helicopter confirm this high speed effect on the loads with the hub bending moments (both flap and lag) being much higher in high speed flight than in hover.

Typically, high speed flight is associated with such conditions as battlefield operations, mercy flights, search and rescue operations (the dash to the search area) and possibly anti-submarine warfare depending on how quickly the helicopter must move from one point to another. The effects of high speed flight may also be exacerbated by high gross weights, bad weather and adverse centre-of-gravity positions which all require more rotor thrust to be produced than would otherwise be needed.

The severity of the damage induced by high speed flight was such that it prompted the Civil Aviation Authority in the United Kingdom to institute the following:

"... *the forward flight speed* [is restricted] *such that the never exceed speed V_{NE} should not be exceeded in the event of a slight upset i.e. if the maximum level flight speed of the helicopter V_H is higher or equal to V_{NE} then a normal maximum operating speed V_{NO} should be imposed which is 10% or 10 kts less than V_{NE} ... This we believe has been a major contribution to helicopter safety.*"[27]

3.4.2 Transitional Flight

Transitional flight is defined as the condition existing when a helicopter is in a flight condition between hover and horizontal flight. It is a relative rather than absolute flight condition and depends on such things as helicopter speed, main rotor speed, and ambient wind strength. Liard[28] states that transitional flight often generates the highest vibratory stresses. Darts and Schutz[15] confirm this with stress histories from a Westland Sea King in which they note that the "... *normal approach to the hover* [is] *one of the manoeuvres that produces the severest loading*". Liard[29] demonstrates the effect that transitional flight can have on the fatigue life of a main rotor blade. Using a helicopter "transport mission" spectrum to give a base life, he compares the lives obtained under other missions. One of these, crop spraying, gives approximately the same life unless prolonged transitional flight is included in the spectrum, in which case the fatigue life becomes less than half the base life.

In terms of frequency of occurrence and severity, the transition from forward flight to hover is the worst type of transitional flying. Whereas pilots will usually move their helicopters quickly from hover to forward flight, the reverse case is different. When a pilot wishes to hover, it is generally in preparation for landing or to hover over a specific spot. Thus the pilot must transition more slowly and therefore spend more time in the transition region with a consequent penalty in the fatigue damage incurred.

The cause of these transitional, vibrational loads is linked to the blade dynamics. A change in flight condition means a change in the required rotor thrust. However, because the blade is a flexible structure which is rotating, cycling in pitch, flapping up and down, leading, lagging, and twisting, any request from the pilot to change state must produce transient responses.

3.4.3 Ground-Air-Ground Cycles

Ground-Air-Ground (GAG) cycles produce significant Low Cycle Fatigue damage to rotor system components[13,22,25]. The GAG cycle is a single load cycle which is applied to each component once per flight. It is made up from the variation in load between the maximum and minimum loads experienced by each component during a flight[24] (Fig. 8). The maximum load will occur sometime during flight, but the minimum load usually occurs on the ground (hence the name, ground-air-ground) where it is either zero or slightly negative*. However, the minimum GAG cycle load may occur during flight if the helicopter enters autorotation mode (see Section 3.4.4).

* A negative ground load occurs, for example, in a helicopter main rotor blade. This comes about because when a main rotor blade is stationary, it droops, thus placing its lower surface in compression.

Krasnowski et al[13] illustrate the effect that GAG load cycles can have on predicted lives. According to them, GAG cycles can cause periodic over-strain in a component. This implies that the High Cycle Fatigue loads can be fatigue damaging despite their being nominally below the material fatigue endurance limit as noted in section 3.3. Werley[25] also shows the effects of GAG cycles in his analysis of a rotor system component failure. In this case, GAG loads measured in service were double those used in the original fatigue qualification of the component. This resulted in the component's design life being reduced from unlimited to 30,000 hours.

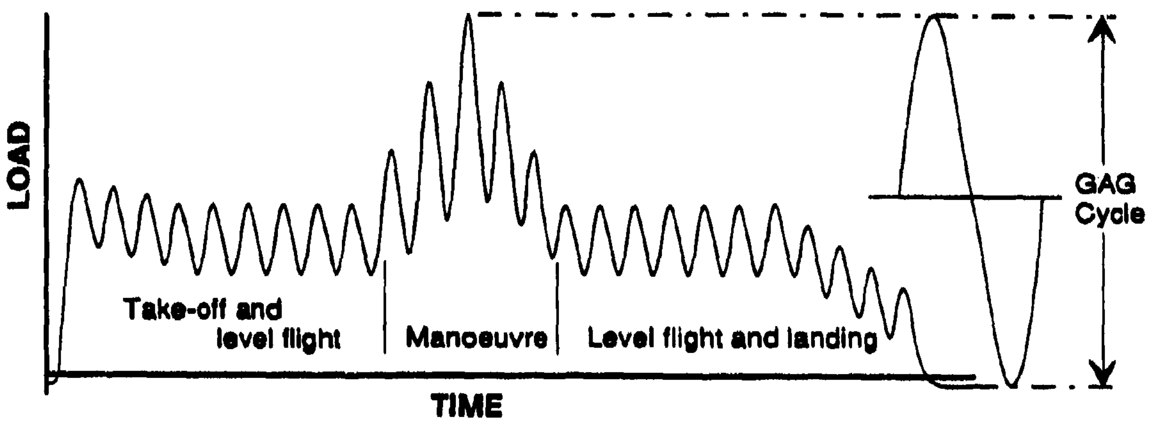

Figure 8: Simplified load time history for a helicopter rotor system component, showing the definition of the Ground-Air-Ground (GAG) cycle.

3.4.4 Autorotation

One of the most severe and most damaging manoeuvres that main helicopter rotors can experience is autorotation.

Autorotation is the condition that a helicopter enters when it loses engine power. This loss of power means that the rotors can no longer generate enough thrust to lift the helicopter and consequently it starts dropping to the ground. In this situation, the helicopter pilot will alter the rotor control settings so that the rotor acts as a windmill, extracting energy from the air and slowing the helicopter descent. The importance of autorotation from a fatigue point of view is that the *"steady load can in some components be completely opposite to the steady* [load] *found for a power on maneuver"*[30]. This reversal results in a GAG cycle being applied to the main rotor system components that is more severe than normal[30]. Two flight surveys, one on a Westland Sea King[31] and one on a Messerschmitt-Bolkow-Blohm BO-105[15] both point out that autorotation was one of the most (if not the most) severe manoeuvres in terms of the loads measured.

Autorotation is particularly relevant to helicopters engaged in the training role since it is a manoeuvre in which pilots must be proficient. In the Sea King survey mentioned in the previous paragraph the authors state that 53% of the flights included at least one autorotation. Also, the total flight time spent in autorotation was 2.5%.

3.4.5 Ground Taxiing

Ground taxiing of a helicopter can lead to fatigue problems in the main rotor head, depending on how the helicopter is taxied. One method of taxiing is for the pilot to tilt the main rotor thrust forward (i.e. apply a large amount of forward cyclic control) until the helicopter moves forward. This method particularly applies to helicopters with wheeled undercarriages and, depending on the softness of the ground, the initial rotor thrust required for taxiing can be large. By using this method, the loads on the rotor system

components will consist of large amplitude cycles which can be very fatigue damaging. A second method of taxiing a helicopter is first to generate enough rotor thrust such that the helicopter almost lifts off and then apply forward cyclic control to initiate the taxi. This leads to higher mean loads than the first method, but cyclic load amplitudes are much reduced. This reduces the fatigue damage caused by the taxiing.

Grainger[32] raises the notion that ground taxiing of a helicopter may induce high loads. In particular, for a helicopter with a semi-rigid head (and a rigid head as well, presumably), *"... small control movements give high rates of response and correspondingly high loads"*.

Symonds[33] mentions that, early in its life, the AH-64 Apache suffered from a series of fatigue cracks in its main rotor hub. The cause of these cracks puzzled the engineers at McDonnell Douglas Helicopter Company (MDHC) until they found out that the Apache pilots preferred to ground taxi their machines by applying full forward cyclic with almost zero collective. In other words, they were using the main rotor to pull the entire weight of the helicopter during the taxiing which produced large cyclic bending moments in the rotor blade support structure and led to the fatigue cracking. To stop this damage occurring, MDHC engineers recommended that the amount of ground taxiing be minimised, but, when ground taxiing was unavoidable, the procedure to be followed should be the same as the second method described at the beginning of this Section.

3.4.6 Droop Stop Pounding

Droop stops are structural elements on the main rotor hub which come into play when the helicopter is at rest. In this condition, the flexible rotor blades sag under their own weight, but the amount of droop is limited by each blade's droop stop. Normally, the blades are well clear of the stops during power-on conditions or the stops themselves are retracted out of the way, but under some flight conditions contact can occur[27,34]. Kieras[34] monitored these contacts on a U.S. Army Black Hawk and found that they occurred at a rate of five seconds per flight hour. The conditions under which contact occurred were: main rotor engagement, backwards taxiing, slope landings and run-on landings when the main rotor was used to brake the aircraft. In general, contact may occur during any ground condition in which the cyclic stick is well off-centre when the collective is reduced. Werley[25] shows that such loads can be large and can cause major reductions in the predicted lives of components.

The danger of droop stop pounding is such that Kieras gives this advice: *"Pilots should avoid DSP* [droop stop pounding] *by using proper technique as pounding of the stops is quite damaging to blade retention components."*[34]

3.4.7 Ground Resonance Damping

Coupling can sometimes occur between the movement of the main rotor centre-of-gravity and the fore and aft or sidewards movement of the main rotor shaft[35]. If insufficient damping is present, then a self-exciting vibration may result with the helicopter rocking from side to side out of phase with the rotor c.g. motion. This phenomenon is called "ground resonance" because the most common occurrence of the condition is on the ground. Here, a helicopter supported by its undercarriage has a low natural frequency in sideways motion, particularly if it has tyres rather than skids. If ground resonance begins, then the resultant coupling of the rotor c.g. motion with the dynamic characteristics of the undercarriage cause a resonance condition which can cause the complete destruction of the helicopter.

The lead/lag hinge is the culprit in this case as it is the presence of this hinge which allows the rotor system to alter its c.g. location (Fig. 9). The condition is prevented or minimised by using a damper on the lead/lag hinge. However, this damper can itself cause problems as it can produce large loads in the rotor system components[25] during landing and cyclic loading during flight as the blade alternately leads and lags during each main rotor revolution.

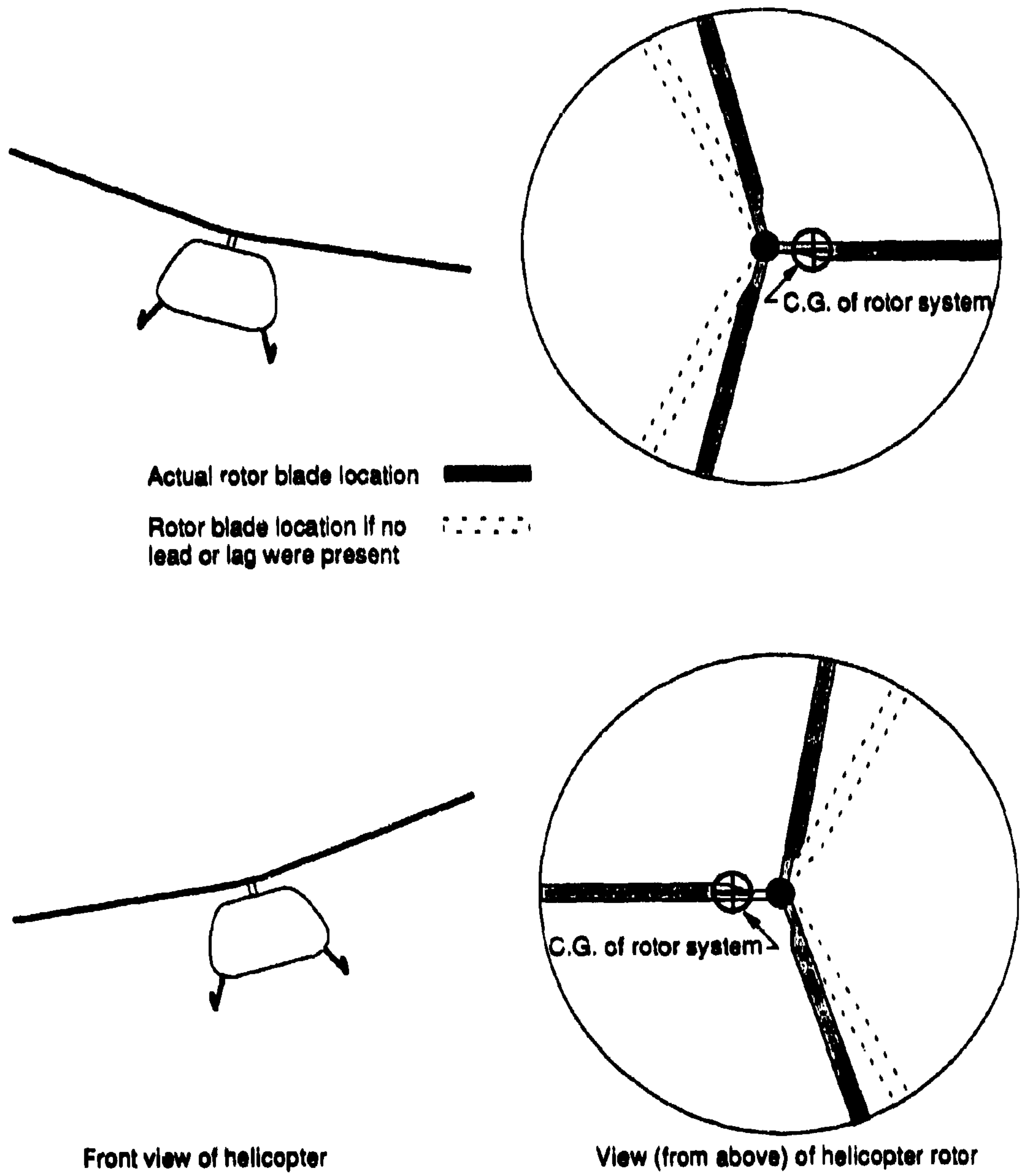

Figure 9: Movement of the rotor c.g. position as the individual rotor blades alternately lead and lag

3.4.8 Operations From Slopes

If a helicopter takes off or lands sideways on a sloping surface then the weight of the vehicle will be temporarily supported by one side of the undercarriage only. According to Crichlow et al[12], this produces a bending moment in the main rotor shaft which the rotor must resist. In their flight tests, on slopes of up to 10°, Crichlow et al measured "... *blade flapping loads which were 80% as great as the maximum encountered in the most severe pitch manoeuvre. However, these high loads lasted for a greater period of time, thus, the peak load count was greater*".

Note that it is not only take-offs and landings from hills that come under this heading. The decks of ships can also be considered as sloping surfaces since, in most sea conditions, ships roll and pitch.

3.4.9 Maritime Operations

Maritime operations include operations from ships and general over-water flying by land-based helicopters (e.g. oil-rig supply helicopters).

Over-water flying may induce fatigue loading if the helicopter is flying in air that is heavy with salt. Kay and Daniels[36] indicate that helicopters operating in the North Sea can experience vibrational rotor loads due to salt accumulation on the blades. On the worst days, they found that salt accumulation could occur at altitudes up to 2000 feet.

Shipboard operations impose severe conditions on helicopters with the main problems appearing in the undercarriage (discussed later). As far as the main rotor is concerned, shipboard operations may have an influence on the rotor loads when the helicopter is flying within the ship airwake. No specific studies have been made into the interaction between main rotor loads and ship airwakes so no definite statements can be made about the effects of this interaction. However, it seems likely that a main rotor, partially or fully inside the turbulent airwake from a ship would be affected. It is possible that the load variations induced could be fatigue damaging.

3.4.10 Pitch Acceleration

Crichlow et al[12] state that the angular acceleration in pitch has an important effect on main rotor loads and offer their flight test data as proof. In the data, they measured high cyclic flap loads under conditions of high pitch acceleration.

3.4.11 Negative Thrust

Another important load case for main rotor components, but not the rotor blades, applies to helicopters which can generate negative thrust from their rotors. This can be in nap-of-the-earth flying which requires negative-g push-overs (e.g. Black Hawk and Apache) or negative thrust during hover (Westland Lynx or Sikorsky S-69 Advancing Blade Concept demonstrator). The reason for the negative hover thrust capability, especially in a naval helicopter such as the Lynx is that, having landed on a ship's deck, the helicopter pilot can use negative thrust to force the helicopter hard against the deck until the ground crew secure the helicopter. By its nature though, the negative hover thrust condition will cause a large ground-air-ground cycle to be applied to some of the rotor system components, and so some navies may elect not to use the capability at all (e.g. the Royal Netherlands Navy[11]).

3.4.12 Blade Sailing

Blade sailing is a term used to describe the behaviour that rotor blades can exhibit during rotor shut down after landing. Rotor blades can be thought of as flexible beams which obtain their required stiffness during flight through the action of centrifugal force. However, after landing, as the rotors slow down, the pilot loses effective control over the blades since their stiffness decreases dramatically with the decrease in rotational speed and

corresponding centrifugal force. In this situation, a sudden gust of wind can cause the blade to make large up and down flapping motions. This can cause the blades to hit the ground and shatter, chop through tailbooms or strike anyone unlucky enough to be inside the rotor radius. From a fatigue point of view, blade sailing can contribute to the GAG cycle by increasing the peak negative loads experienced by a rotor blade. Blade sailing is more likely to occur if the rotor blade construction is very flexible and/or the rotor head uses hinges to allow flapping motion.

3.5 TAIL ROTOR LOADS

The tail rotor may be viewed as a scaled down version of the main rotor and, in many respects, the generation of loads is the same. The main differences are that the tail rotor rotates at a much higher speed than the main rotor: 15 – 30 Hz versus 3 – 6 Hz respectively, and that the magnitudes of the loads (but not the stresses) are much smaller. Hence, the tail rotor experiences more high cycle fatigue load cycles than the main rotor.

The most critical fatigue regimes for a tail rotor are the hover, low speed flight, rearwards flight and sideways flight[27,32]. This is because under these conditions the tail rotor is providing all or most of the force required to counteract the torque of the main rotor. Also, rearwards flight can generate high tail loads, especially if the helicopter is unstable in this flight mode. As forward speed increases, the engine power required to fly the helicopter decreases rapidly before reaching a minimum, generally near its maximum endurance speed, and then begins to increase again (Fig. 10). Therefore, the tail rotor thrust required to counteract the main rotor torque decreases with decrease in engine power. In addition, fins mounted on the helicopter tailboom begin to generate significant aerodynamic forces which are used to aid the tail rotor in its anti-torque work. Consequently, the tail rotor thrust required is again reduced. In helicopter terminology, the tail rotor is said to be unloading. As speeds rise further and the engine power required increases, the tail rotor thrust must also increase, but due to the presence of the fins the anti-torque thrust required from the tail rotor is not as large as at low speeds.

Grainger[32] cites amplitude and rate of yaw as being the most critical parameters though offers no evidence to support this. High rates of yaw and/or large yaw angles are accompanied by large changes in tail rotor loads and the larger the change in load, the greater the fatigue damage caused. McCue et al[37] in their examination of air-to-air combat roles for helicopters also make the observation that high yaw rates and sideflare manoeuvres can generate large loads on the tail rotor.

Manoeuvres which correspond to these conditions include rapid turns, turns with abrupt stops, steady sideslip, tail rotor pedal reversals, and sideways flight. Rearwards flight might also be included if the helicopter is unstable when flying backwards. Maltby and Hicks[31] note that the Westland Sea King does have this instability and therefore in rearwards flight, "*... high rates of turn in the yaw axis can easily be generated. Large tail rotor loads might be anticipated in these circumstances*". The conditions which correspond to these manoeuvres include landing, search and rescue (low speed manoeuvring to effect a rescue), anti-submarine warfare (during loitering while listening to sonobuoy signals), battlefield operations, and crop dusting.

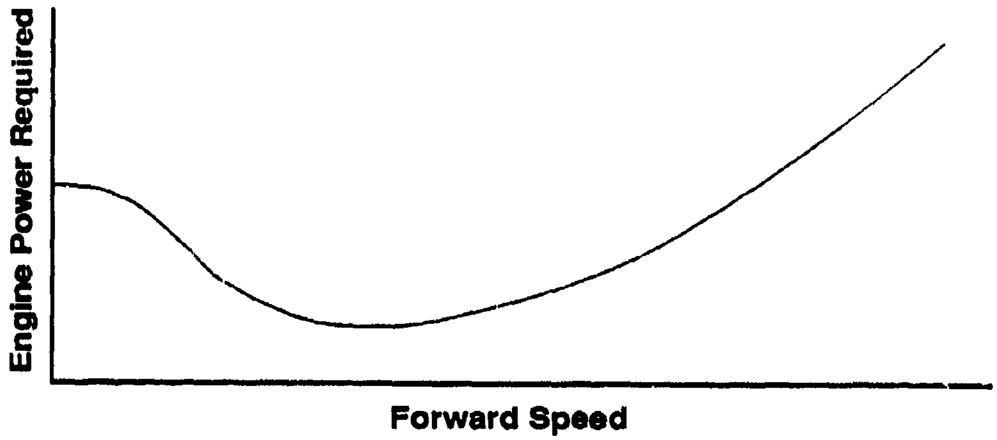

Figure 10: Engine power requirements versus forward flight speed

3.6 AIRFRAME

The airframe consists of all the helicopter structure which is not part of the engines, mechanical drive system, rotor system or flight control system, that is, the fuselage, undercarriage and auxiliary aerodynamic surfaces such as tailplanes.

Though the airframe is the part of the helicopter most like a fixed-wing aircraft, its fatigue-critical load cases are not necessarily the same. This arises because of the different ways in which lift is produced. Whereas lift is an important parameter affecting airframe fatigue in fixed-wing aircraft, main rotor lift is largely irrelevant to a helicopter airframe. By its design, a helicopter airframe does not experience the lift forces produced by the main rotor. Therefore, airframe fatigue is less of a problem for helicopters than fixed-wing aircraft, but it is not necessarily unimportant. One incident, related to the author of this report, concerned the failure of a side door on a Westland Lynx. Although not structurally important in itself, when the door came off, it struck the tailboom, resulting in the loss of the helicopter.

3.6.1 Main Rotor Airframe Attachment

In order for the main rotor to lift the airframe, a load path must be provided between them. On some helicopters this is achieved by mounting points on the main rotor gearbox casing attached to the fuselage lift frames. In this situation, the helicopter literally "hangs-off" the main rotor shaft. Canedale[10] refers to the attachments forming such a load path as the *"airframe vital parts"* in that a failure here can have disastrous consequences.

The loading on the airframe/main rotor attachments is mostly due to the main rotor loads and is therefore dynamic in nature. This loading occurs at the blade passing frequency* and its harmonics[9,38,39,40,41]. It consists of vertical and rolling/pitching vibrations due to the rotor blade flapwise and chordwise shear forces, bending moments and torsion moments (see, particularly, Ref. 42, pp 312-319). The lift frames can experience fatigue damage from such loading. Conditions which aggravate this damage include adverse locations of the c.g. and operations at the maximum allowable gross weight. In combination, these two conditions (i.e. a fully loaded helicopter operating at the extremes of its c.g. envelope) can cause the structure to crack[42].

* If a helicopter rotor has b blades and is rotating at a frequency Ω, then its blade passing frequency is $b\Omega$. In simple terms, the blade passing frequency indicates how often a point under the rotor sees a blade pass overhead.

McDonnell Douglas Helicopter Company uses a "static-mast"[83] to connect the main rotor to the fuselage (Fig. 11). In this design, a hollow tube (the mast) is rigidly mounted to the helicopter fuselage and the main rotor drive shaft passes through it to turn the main rotor. By the use of suitable thrust bearings, the mast is made to carry all the rotor lift and horizontal thrust while the drive shaft only has to transmit the engine torque.. The "static" part of the "static-mast" name is derived from the fact that the horizontal rotor thrust now acts as a static bending load on the mast instead of a cyclic load on the drive shaft.

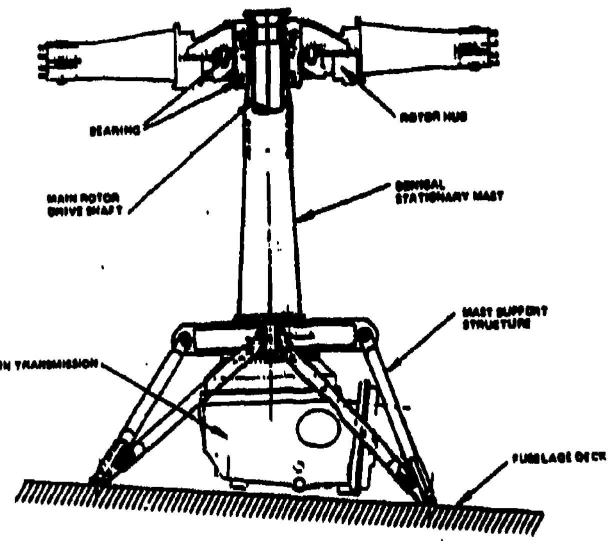

Figure 11: The McDonnell Douglas "Static-Mast" mounting system for the Apache main rotor.

3.6.2 Slung Loads

A slung load is a load carried externally and suspended under the helicopter from one or more cables. Generally, it is suspended from a single point via straps or cables. Slung loads affect the fatigue loads on the helicopter because they have the potential to produce transient and dynamic cable loads which are larger than the actual weight of the slung load.

Transient strap loads are generated during take-off when the cable takes the strain of the load. Battersby[43] describes the results of tests in which the maximum values of these transient loads were measured. The ratio of dynamic to static loads varied from 1.0 to 1.23 in all the tests with the exception of a single value at 1.7*. Therefore, the size of the dynamic load cycle applied to the helicopter load lifting points could be significant if the helicopter is often made to lift the maximum allowable slung load. The main factor which influences the size of the transients is pilot technique. A pilot who has not had much experience at lifting slung loads may not perform the take-off as smoothly as an experienced pilot. Battersby makes a special note that the helicopters used in the tests in his report were flown by an experienced test pilot. The type of material used in the straps was not a factor in the magnitude of the transient loads measured in these tests.

Dynamic strap loads are experienced during flight and are the result of the load oscillating like a pendulum at the end of its support straps. Dynamic to static load ratios of 1.0 to 1.25 were measured by Battersby for level flight speeds ranging from 30 kn to 90 kn. The frequency of the dynamic loading was close to the frequency of rotation of the main rotor. Again, the caveat applies that these results were obtained with an experienced test pilot flying the helicopter. The possibility exists that an inexperienced pilot may induce larger load oscillations and therefore more fatigue damage to the load lifting points. An important factor is the density and aerodynamic drag of the load to be carried[43,44] with

* The ratio of dynamic to static was calculated by taking the maximum measured transient load and dividing it by the weight of the slung load.

Battersby finding that high density, low drag bodies tended to oscillate while low density, high drag loads maintained an almost constant trailing angle under the helicopter. Another important factor is the size of the load and its distance from the main rotor. Large loads carried close under the helicopter developed significant dynamic loading due to the main rotor downwash.

There is also the possibility that the load may become aerodynamically unstable if the helicopter is travelling too fast. This would induce high loads, not only in the airframe but also in the main and tail rotors as the pilot fought to regain control. In this case though, it is expected that a pilot would jettison such a load before the helicopter became uncontrollable and suffered damage.

A special case of the slung load that merits mention is the "towed load". In this case, the helicopter is towing a load that is either on or under water. Towing through water will generate higher static tension loads in the tow cable than if the load were simply being carried through the air. The effect on the dynamic loads is not as clear however, because while the oscillatory motion of the load will be dampened by the water, wave motion may exacerbate it. The towed load case can be critical for the main rotor drive shaft as the extra horizontal thrust means increased bending stresses on the shaft, unless the helicopter has a "static-mast" or similar (see Section 3.6.1).

3.6.3 Tailboom Loads

Thrust produced by a tail rotor counteracts the tendency of the fuselage to spin in the opposite direction to the main rotor. Since the tail rotor is located at the end of a tail boom which can normally be considered as long and slender, the magnitude of the bending moments produced is important. As well, since tail rotors are generally mounted on a fin extending from the tail boom, torsional loading may also exist in the tail boom (Fig. 12).

Lamb et al[45] measured the magnitude of the bending and torsional moments produced in the tail boom of an SH-60B Seahawk and concluded that *"... fatigue and peak limit loads on dynamically loaded components* [were reached] *before any of the airframe DNE* values were reached"*. This does not mean that the airframe loads were not fatigue damaging, just that they never exceeded the maximum design loads. However, it does indicate that in normal flight, the fatigue damage on the fuselage due to tail rotor loads in the Seahawk and Black Hawk is of lesser importance than the fatigue damage which occurs in rotor system components.

Conditions which increase the required tail rotor loads are, as indicated in Section 3.5, the magnitude and rate of yaw. Similarly, the aggravating manoeuvres are rapid turns, turns with abrupt stops, tail rotor pedal reversals, steady sideslip, sideways and rearwards flight, and sideflares.

Tailboom inertial loads are a consequence of having a large mass (tail rotor and gearbox) mounted at the end of the tailboom. When the helicopter undergoes rapid pitch or yaw accelerations, large inertial loads can be generated along the tailboom. For example, the AS350 Squirrel has a very light tailboom and at least one Australian Army Squirrel, used for pilot training, has had its tailboom destroyed by buckling during a heavy landing.

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* DNE: Do Not Exceed, i.e. maximum allowable loads

Figure 12: Photograph of a Royal Australian Navy S-70B-2 Seahawk coming in to land. The skin of the tailboom shows tension field buckling caused by the torque loading of the tail rotor. (Photograph courtesy of Aerospace Technologies of Australia)

3.6.4 Auxiliary Aerodynamic Surfaces

Most helicopters employ auxiliary aerodynamic surfaces attached to the airframe as aids in forward flight (Fig. 13).

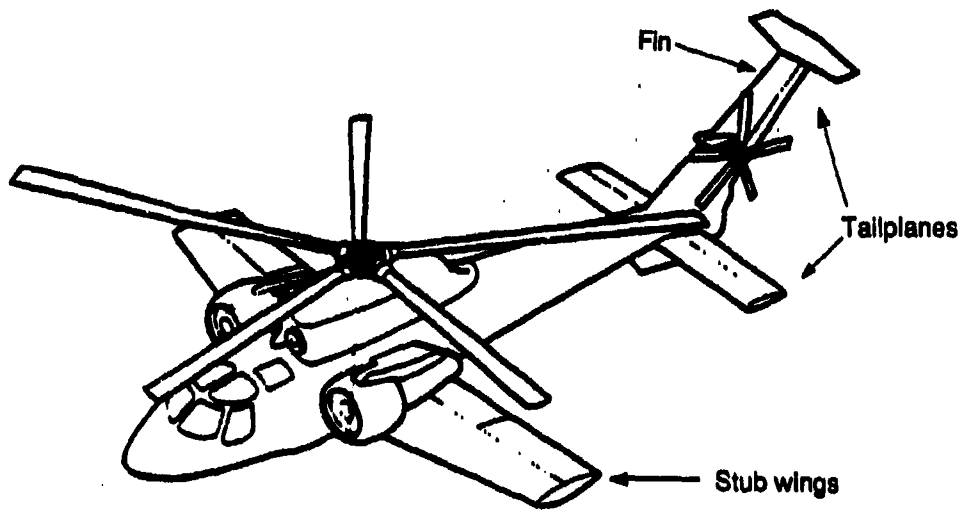


Figure 13: The Rotor Systems Research Aircraft (used by NASA for research into rotary-wing aircraft) showing off its array of auxiliary aerodynamic surfaces.

(a) Fins are used to provide assistance to the tail rotor in producing an anti-torque force (See Section 3.5). As forward flight speed increases, the fins will provide an increasing anti-torque force. Only the fin and the fin/tailboom attachment structure will see large changes in load as the fin gradually takes over the function of the tail rotor. Other parts of the airframe see the combined loading from tail rotor and fins and would therefore not experience any large load changes.

(b) Tailplanes and stabilators are used to trim the helicopter in forward flight, thus relieving the main rotor of this task. The difference between the two is the same as for fixed-wing aircraft, i.e. a stabilator is simply an all-moving tailplane. The significant load cases, from a fatigue standpoint, depend mainly on where the tailplane or stabilator is located with respect to the main rotor. Also, the significant fatigue load cases for a tailplane may differ from those for a stabilator because of the stabilator's all-moving nature. Cracking in the tailboom attachment fittings is the most common form of fatigue failure to occur in tailplanes and stabilators.

For example, consider the case of a helicopter which has entered autorotation. On a tailplane-equipped helicopter, the tailplane will be producing a tail-up moment whereas just prior to autorotation it would have been producing a tail-down moment. The severity of the change (tail-down to tail-up) is highly dependent on the skills and knowledge of the helicopter designer in much the same way that the severity of the stall is in fixed-wing aircraft. If, on the other hand, the helicopter is equipped with a stabilator then this will be under the control of the automatic flight control system (AFCS). Therefore, the actual stabilator loads existing before and after autorotation, and hence the change in load, will be dependent on the operational parameters of the control system.

The other important flight conditions, as already stated, depend on the location of the tailplane. Prouty[38] compares and comments on the four most common configurations and these are shown in Fig. 14. Consider the basic configuration shown in Fig. 14(a). In the hover, the tailplane is well clear of the main rotor wake, but as the helicopter transitions from hover to forward flight, the rotor wake begins to impinge upon and finally envelop the tailplane. This causes a sudden nose-up pitching motion which requires the pilot to take corrective action. As Prouty points out, this was fine as long as the tailplanes were small in area and disc loadings were low. Nowadays though, high disc loadings and large tailplanes are common and thus the potential for large forces to be generated during the transition is high, which means that this configuration is no longer in favour.

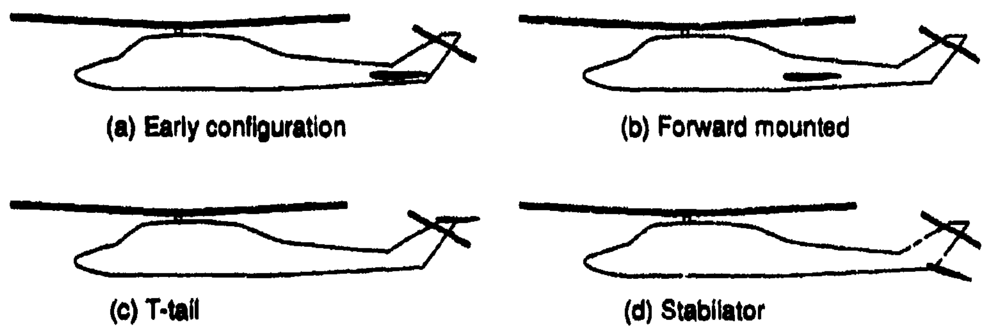


Figure 14: Typical locations for tailplanes and stabilators

Figure 14(b) shows the solution adopted by engineers at the Bell Helicopter Textron Company. In this case, the tailplane is far enough forward such that it is always inside the wake and therefore there is no transition to worry about. The third configuration shown in Fig. 14(c) is known as the T-tail and is used on many helicopters (e.g.: Westland Sea King and Lynx and the Sikorsky CH-53 Sea Stallion). This removes the main rotor wake interference in most flight conditions. However, unless the tailplane is mounted high enough, the main rotor wake will impinge upon it at high speeds. This causes turbulence which can induce severe vibratory loads. The fourth, and final, configuration (Fig. 14(d)) considered here consists of an all-moving tailplane mounted at the end of the tailboom and controlled by an AFCS (e.g.: Sikorsky Seahawk and Black Hawk, McDonnell-Douglas AH-64 Apache). The AFCS has the task of ensuring that the tailplane is always aligned with the local airflow such that sudden changes in load do not occur. Unfortunately, control systems are not perfect and under some extraordinary combination of circumstances they can induce loads which are detrimental to the helicopter. This applies to all helicopters with an AFCS installed. For example, Maltby and Hicks[31] in analysing the flight loads from an AFCS equipped Sea King, note that an AFCS can sometimes induce "*... severe oscillatory loadings ... in response to apparently innocuous actions by the pilot*".

(c) Stub wings are small wings attached to the side of a helicopter fuselage near the main rotor shaft. Originally, they were used to assist the main rotor by relieving it of the need to provide all the lift in cruising flight. Perhaps the best example of this that is still operational is the Mil Mi6 Hook (Fig. 15). Developed in the U.S.S.R. in the late 1950s, the wings of this helicopter produce a significant percentage of the required lift when cruising[5,8].

Nowadays, stub wings are mainly used as convenient places from which to hang stores (e.g. AH-64, UH-60 with ESSS - External Stores Support System (Fig. 16)) and any aerodynamic lift that they produce is incidental.

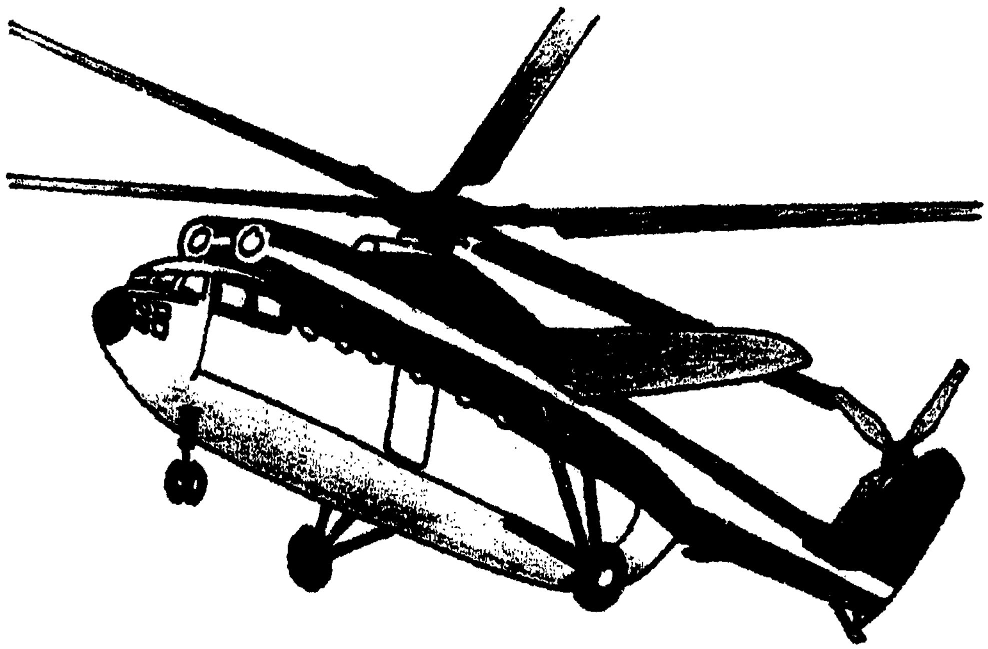


Figure 15: Russian Mil Mi6 "Hook" helicopter showing the stub wings attached to its fuselage

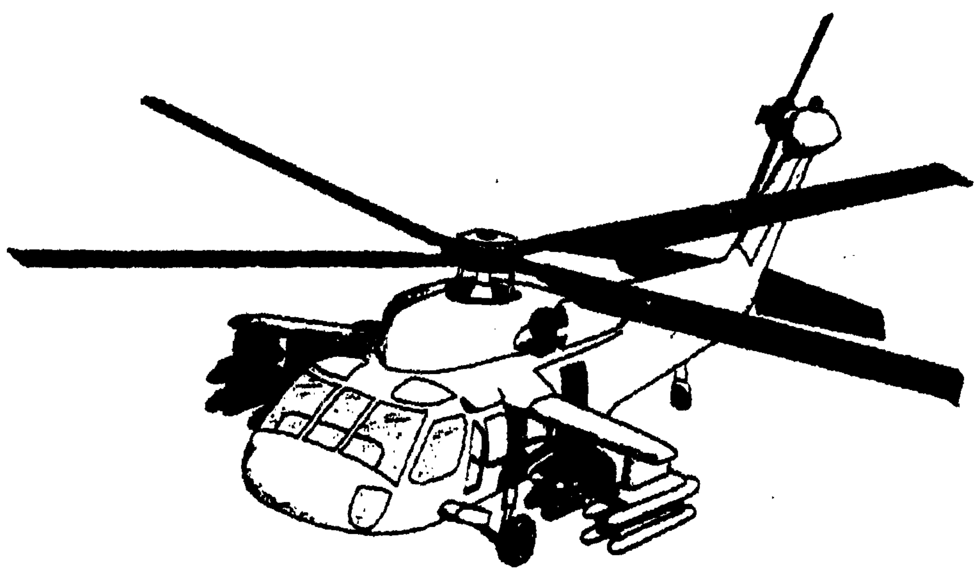


Figure 16: Sikorsky UH-60 Black Hawk with ESSS (External Stores Support System)

The types of loads applied to the helicopter fuselage by the wings will depend on the helicopter speed. Under normal cruise or higher speed flight, the wings may be considered to be equivalent to those of a fixed-wing aircraft and therefore treated the same. At low speeds, the situation is very different as the wings no longer produce any appreciable lift. Instead, they present a large flat plate area to the main rotor downflow which impedes the ability of the rotor to generate lift*. In this case, the magnitude of the loads would

* In fact, the wings of the Mi6 are removed if the aircraft is operated in low speed roles, e.g. as a "flying crane".

probably be less than under higher speed flight, but they would be oscillatory with a frequency equal to the blade passing frequency.

3.6.5 Undercarriage Loads

Helicopter undercarriages consist either of skids (with or without oleos) fixed to the fuselage, or wheels connected to the fuselage by oleos (Fig. 17). According to Grainger[32], the driving force behind helicopter undercarriage design is increased energy absorption. This aim is the same as that required for the design of naval (fixed-wing) aircraft which must operate from aircraft carriers. The Black Hawk undercarriage, for example, is designed for operational rates of descent of up to 9 feet/second and the Seahawk at 12 feet/second[46]. Compare this to the U.S. Navy F/A-18 Hornet which has landing gear designed for landings at up to 24.8 feet/second vertical descent rate[47].

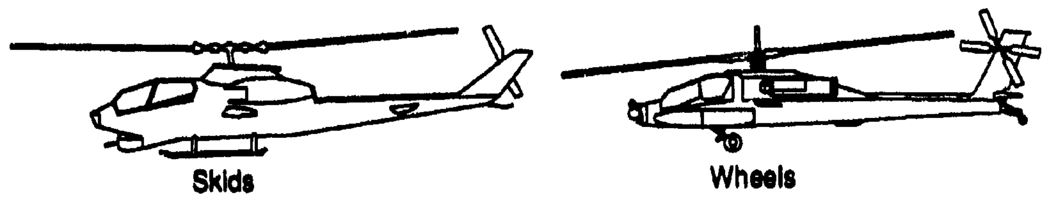


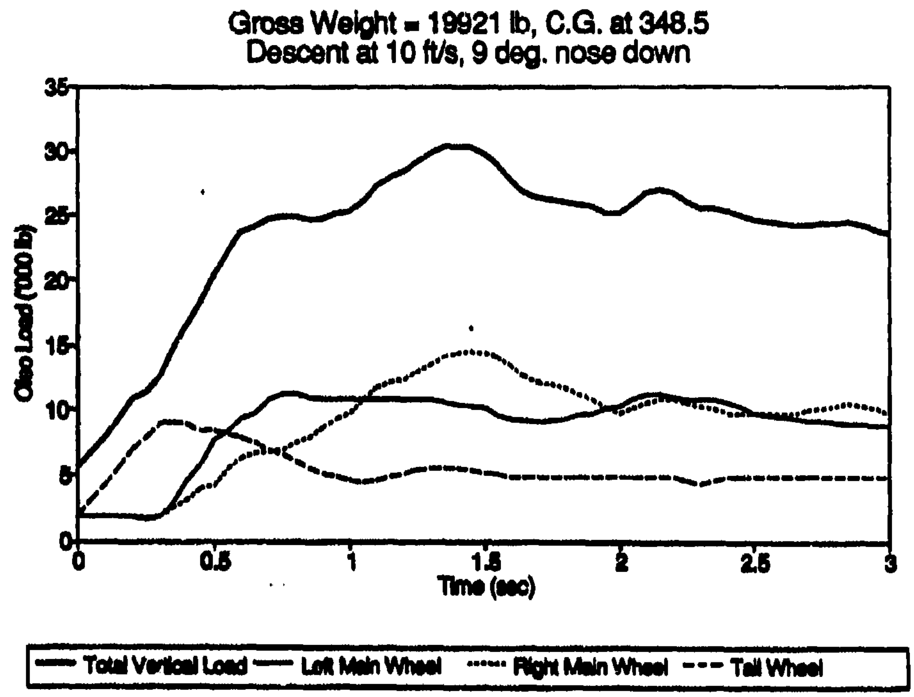
Figure 17: The two types of helicopter undercarriages

A helicopter undercarriage is acted upon by three loads: vertical, side and drag loads[14], with the vertical load being the largest of the three. Time histories of the undercarriage oleo loads measured in a U.S. Navy helicopter are shown in Fig. 18 for two different sets of landing conditions : fast descent, nose down and slower descent, level landing. As seen in the time history for the high speed, nose down landing, the combined total of the vertical loads may sometimes significantly exceed the gross weight of the helicopter during landing.

For both types of undercarriage, drag loads arise during landing if the helicopter makes a run-on landing, i.e. it lands with some forward velocity. For helicopters with wheeled undercarriages, drag loads can also arise during such ground operations as taxiing, turning and braking. Side forces are generated under similar conditions, i.e. landing with a sideways velocity and during ground operations. As well, significant side loads occur if the helicopter lands or takes off from a sloping surface (i.e. hills or ships).

The three forces are independent of each other in so far as they do not each reach their maximum at the same or even related times[14]. The magnitude of the vertical load will depend on the main rotor lift* and the rate of descent; the drag load will depend on the vertical load, any forward/rearward velocity, ground friction and the amount of wheel braking applied while the side load will depend on the vertical load, any sideways velocity and ground friction. As well all three will depend on the softness of the take-off/landing area and the stiffness characteristics of the undercarriage system.

* In the case of the Westland Lynx and Sikorsky S-69, the ability to produce negative rotor thrust is also a factor to be taken into account. The rotor of the S-69 can produce negative thrust equal to 20% of the maximum gross weight. During landings in the S-69 the pilots would reduce the rotor lift to zero as it touched down and then apply full negative thrust. This procedure was soon abandoned because it was found that the seals in the landing gear oleos were blowing out.



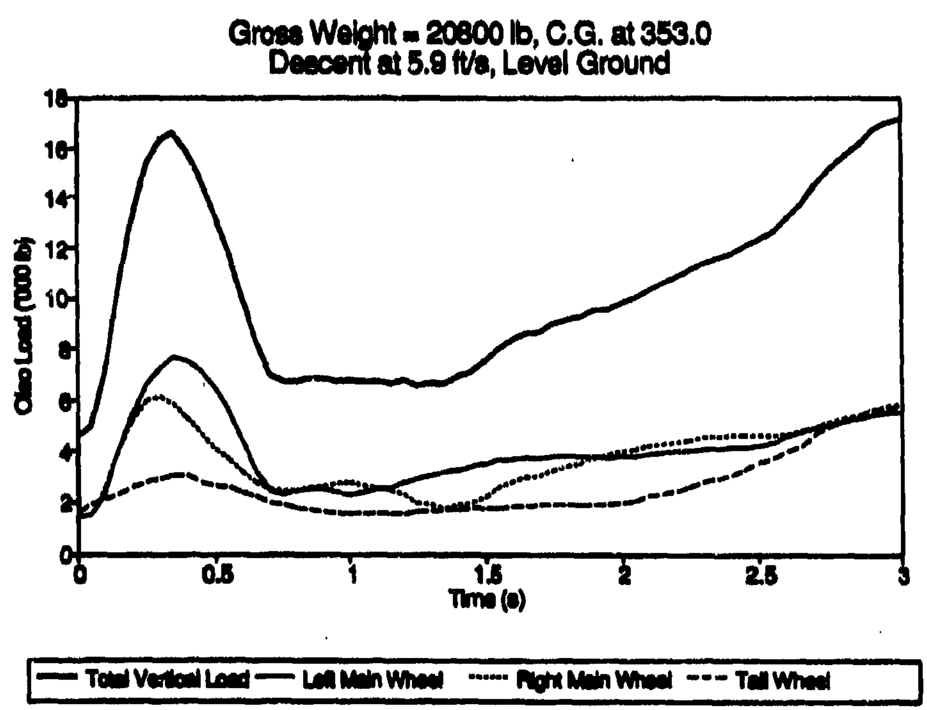


Figure 18: Load-Time history showing the loads measured in landing gear oleos during two different landing conditions (data obtained from tests on a U.S. Navy helicopter).

The worst situations for undercarriage loads are:

(a) Autorotation - A landing during autorotation involves vertical descent rates that are much higher than normal. Therefore, the loads on the undercarriage are likely to be fatigue, if not structurally, damaging. If the helicopter is used in the training role, and it is cleared to routinely perform autorotative landings, then the chance of significant fatigue damage occurring is high.

(b) Slope Landings and Take-offs - Landing or taking off from sloping surfaces means that one side of the undercarriage must carry large bending moments. Black Hawks and Seahawks are allowed to land with a maximum descent speed of 6 feet/second on sloping surfaces; a descent rate which is much less than that allowed for a flat surface[46].

(c) Shipboard Landings - Operating from ships, especially small ships, can impose severe loading on helicopter undercarriages just as it does on naval aircraft. The reason is that a ship is not a conveniently stationary operating platform, but is a rolling, pitching, yawing, heaving, swaying and surging nightmare for pilots. In fact, helicopters operating from ships are cleared to take off and land only during specified weather conditions. The better the performance and agility of the helicopter and the stronger its structure, the larger the operational weather envelope becomes. In addition, because large ships (e.g. aircraft carriers) are more stable in the water, operations from them are less restrictive than operations from small ships such as destroyers and frigates.

(d) Assisted Landings - There are several systems in use to assist the landing of a helicopter on a ship. The Seahawk, for example, has an option to be fitted with a system called RAST (Recovery, Assist, Secure, and Traverse). When the Seahawk is ready to land, the pilot hovers over the landing deck and lowers a cable to the ship which is then used to haul up the RAST cable. Once secured to the helicopter, the RAST cable is then tensioned in order bring the helicopter to a steady hover above the landing point. Having steadied the helicopter, the cable load is then increased to bring the Seahawk in to land[49]. According to a Sikorsky report[48], a RAST landing can produce significant nose-down pitch rates combined with forward motion when the Seahawk tailwheel hits the deck*. This may induce significant loads in the undercarriage.

3.6.6 Tie-Down Loads

Helicopters on ships are usually tied down by cables, chains, or straps. By their nature, cables, chains, and straps can only carry tensile loads and so, as the ship rolls and pitches, they might periodically lose their tension. The actual loads induced in the tie down cables will vary, depending on the sprung mass of the helicopter**, ship velocities and accelerations (e.g. heave, pitch, roll, etc), the amount of tension initially applied to the cables and, possibly, the cable material.

These tie-down loads will be supported by the airframe and, depending on the flexibility of the helicopter fuselage, may be significant in terms of fatigue damage. Unfortunately, the magnitude of these loads is not known by the author of this paper since an extensive literature search failed to turn up even one report or article dealing with the measurement of these loads.

3.6.7 Ground-Air-Ground Cycles

Just as for rotor system components (See Section 3.4.3), Ground-Air-Ground cycles can be fatigue damaging, particularly to the lift frames[39].

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* According to this report, the normal landing method for a Seahawk is that the tail wheel touches down first, followed by the left main wheel and then the right main wheel.

** The sprung mass of a helicopter is the mass supported by its undercarriage.

3.6.8 Panel Resonance

The panels in a helicopter fuselage can be susceptible to structural resonances due to the high frequency loading transmitted by both the main and tail rotors[32]. Structural resonance implies the application of a large number of low magnitude load cycles to the panels and their fasteners. Tail booms can be particularly susceptible to resonance problems since they are generally long and slender structures.

3.6.9 Main Rotor Wake/Fuselage Interactions

This area has only recently begun receiving more than a cursory glance from researchers[19,20,21] and it remains to be seen whether loads generated during such interactions are significant or not. The report by Battersby[43] cited in section 3.6.2 indicated that such interactions were important for slung loads carried on a cable, close under a helicopter fuselage.

Researchers at the University of Maryland[20] in the U.S.A. performed an experiment in which they measured the unsteady pressures induced on the fuselage of a generic helicopter model with a four-bladed main rotor. They noted that the corresponding unsteady loads on the fuselage (lift, side force and pitching moment) are significant relative to the mean wake interaction loads seen by the fuselage. The scope of this study though, did not extend as far as attempting to compare these loads with the loads seen by a normal helicopter in flight.

It is interesting to note that researchers in a related field, wind turbine research, consider that the interference between the turbine blades and their support tower to be an "*... important fluctuating load source*"[50]. Also, one NASA report[51] describes a wind turbine which was constructed with an open truss-style tower in an effort to minimise turbulent airflow on the rotor and turbulent loads on the tower.

4. FATIGUE DESIGN

4.1 TRADITIONAL FATIGUE DESIGN METHODOLOGY

Helicopter manufacturers use what is known as the "safe-life" approach when determining the fatigue lives of components. This approach has three key elements:[52,53,54,55]

(i) Material and component fatigue data (i.e. S-N curves)

(ii) Load spectrum

(iii) Damage hypothesis and cycle counting (relating the load spectrum and the S-N curve)

In essence, the philosophy of the safe-life approach is to reduce the chance of a failure within the design life of a component* to an insignificant level[26].

The safe-life approach is shown diagrammatically in Fig. 19. Variants of this figure are common in papers on helicopter fatigue (see, for example, Refs 53, 26, 56, and 57), but, as Noback[53] says, this may "... *suggest that everything is clear and that indeed a well-established methodology for helicopters exists. This is not exactly true*".

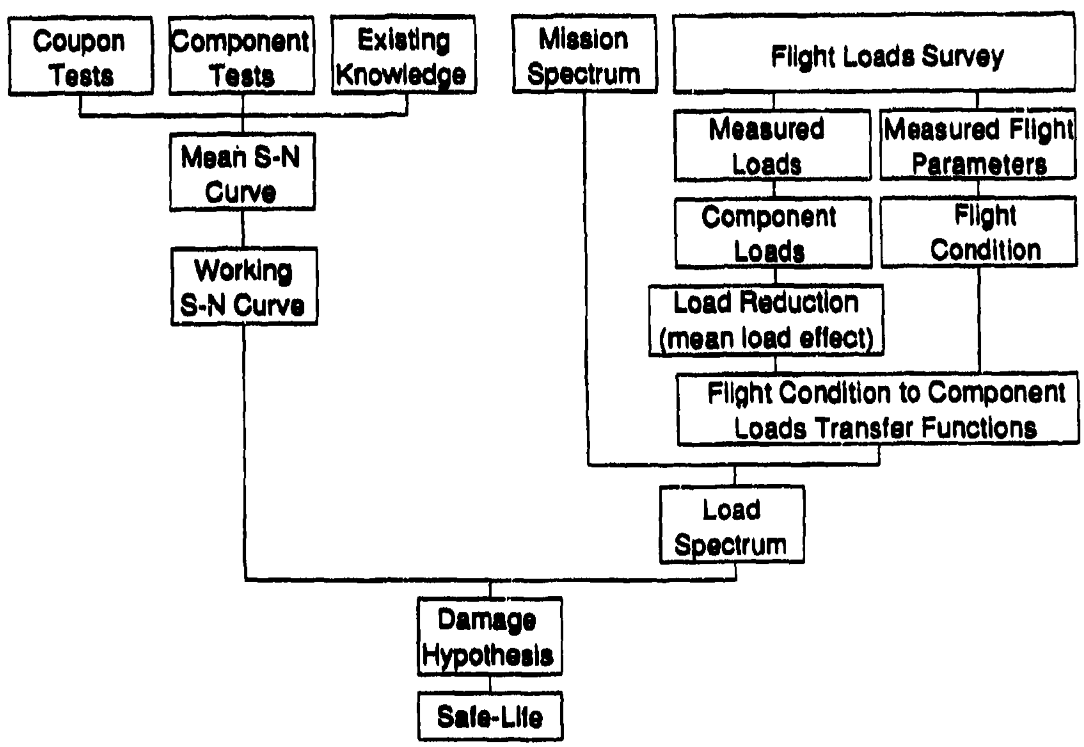

Figure 19: The safe-life approach to fatigue design

A case which illustrates the disparities in the methodology is the hypothetical fatigue life evaluation exercise[58], presented by the American Helicopter Society to several helicopter manufacturers in 1980. The aim of the problem was for the manufacturers to determine the life of a typical helicopter component, in this case a rotor-head pitch-link. Each manufacturer was provided with the same information and asked to determine two lives for the pitch-link: one based on the manufacturer's preferred method of load spectrum

* The definition of component used here includes everything from simple links to large sub-assemblies such as tailbooms.

cycle counting and the other using the block counting method*. The results[59] are presented in Table 3.

Table 3

Summary of calculated fatigue lives (in hours) for the hypothetical pitch-link exercise

Manufacturer	Block Counting Method	Manufacturer's preferred cycle counting method
Aerospatiale[a]	9	58
Agusta	804	6450
Bell[b]	1831	27816
Boeing-Vertol	1294	22523
Hughes[c]	2594	24570
Kaman	861	5901
Sikorsky	240	470

Notes: [a] Now part of Eurocopter
[b] Now Bell Helicopter Textron
[c] Now McDonnell Douglas Helicopter Company

All the analyses followed the procedure of Fig. 19. As can be seen, the lives varied from 9 to 2594 hours for the block counting method and 58 to 27816 for the cycle counting method.

To go into all the aspects of the safe-life method in minute detail is beyond the scope of this report, so what follows is an overview of the fatigue design techniques used by the major helicopter manufacturers. For more detailed information, the reader is referred to the design guide[60] issued by the Advisory Group for Aerospace Research and Development. This 278-page document, published in 1983, contains an extensive survey and evaluation of traditional helicopter fatigue design procedures.

4.1.1 The Working S-N Curve

To calculate fatigue lives, some relationship must be obtained between cyclic loads and the corresponding fatigue damage produced for a particular component. In other words, an S-N** curve (or curves) needs to be constructed for each component. This S-N curve is based both on coupon and component tests, as well as the accumulated knowledge of each manufacturer. During the initial design phase of a new helicopter, components are not available so the designer makes do without component tests. In the design validation phase, components are available and are tested to validate the S-N data used. Examples of such tests, performed on several components of the Apache, are given by Deveaux et al[61]. Component tests are done on small numbers of components (usually not more than six[53,62,63]) because of time and cost constraints. In the Deveaux Apache tests, for example, the number of components tested varied from one to three. Components are tested under spectrum loading, block loading or S-N loading (i.e. constant amplitude loading). S-N testing tends to predominate[24,53,64] mainly because of the "... *difficulty* [in] *predicting the dynamic loads prior to flight testing* ...".

What happens next is, nowadays, accomplished by computer, but the manual graphical method of solution is presented as it best describes the methods used.

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* The block counting method is also known as the peak cycle counting method. This method, rather than counting each cycle in a manoeuvre, assumes that the largest load cycle in the manoeuvre acts for the entire duration of that manoeuvre (See Section 4.1.4)

** An S-N curve is one which defines the relationship between the endurance, N, of a material as a function of the applied stress or load amplitude, S. N is given in terms of the number of load cycles endured prior to failure at a given value of S.

The results of the component tests are plotted on an S-N diagram and individual S-N curves drawn through each test point. The curve shape is normally derived from coupon tests on the same type of material and under the same type of notch and fretting conditions[53]. The coupon test curve shapes are either defined graphically or mathematically. Reference 60 (p116) indicates that a commonly used equation to represent the S-N curve from N = 1 (static strength S_u), to N = ∞ (the endurance limit, S_{∞}) is that proposed by Weibull in 1961:

$$S = S_{\infty} + (S_u - S_{\infty})e^{-\alpha(\log N)^{\beta}}$$

where the values of the constants S_u, S_{∞}, α, and β are obtained from the test data by a least squares fit. This equation and its many variations can be found in the literature (eg Refs 29, 57, 65, & 85) along with the constants used to define the curve shapes. These constants vary widely from manufacturer to manufacturer.

The several S-N curves which arise when the standard curve shapes are drawn through the test points are then consolidated into one curve which is the statistical mean of all the curves (Fig. 20). This is the "Mean S-N Curve" mentioned in Fig. 19.

The next step is to produce the Working S-N curve (i.e. the curve used in the actual fatigue life calculations) by applying a reduction to the Mean S-N curve. The amount and type of reduction vary from manufacturer to manufacturer, though the underlying principles are similar[53,60].

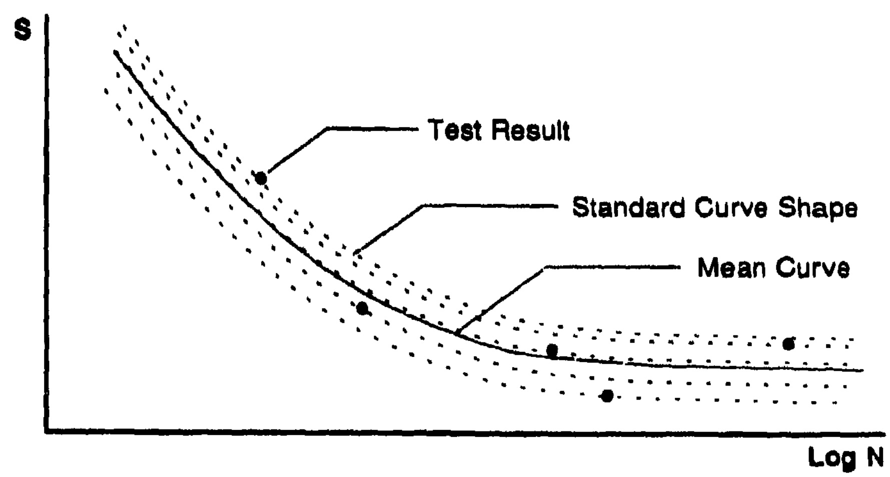

Figure 20: Derivation of the Mean S-N Curve from the test results

At the McDonnell Douglas Helicopter Company (MDHC), for example, the Working S-N curve is initially considered to be the least of the following:[66]

- The mean curve reduced by three standard deviations
- 80% of the mean value curve
- The S-N curve drawn through the points of lowest life.

If less than six component test results are available, then MDHC reduces the Working S-N curve further by the application of a constant reduction factor. This factor is either 0.73, 0.83, 0.9, or 0.96 depending on whether two, three, four, or five components, respectively, are tested. These reductions apply to the Mean S-N curve in the high-cycle region which is defined as the region where, approximately, N > 10^4 or 10^5 cycles.

In the low-cycle regime, other reductions are used. At 10^4 cycles, MDHC shifts the Mean S-N curve down by 1.5 standard deviations; a figure which they base on experience. Similarly, they also assume from experience that the reduction at 10^2 cycles is 10% and at 0.25 cycles* it is 7%. Another common type of reduction in the low-cycle regime is to apply a safety factor to the life (i.e. the number of cycles, N), rather than the load, S as MDHC does. Sikorsky, for example used factors of between 5 and 10 during design of the Black Hawk. Ultimately, a Working S-N curve is produced as shown in Fig. 21.

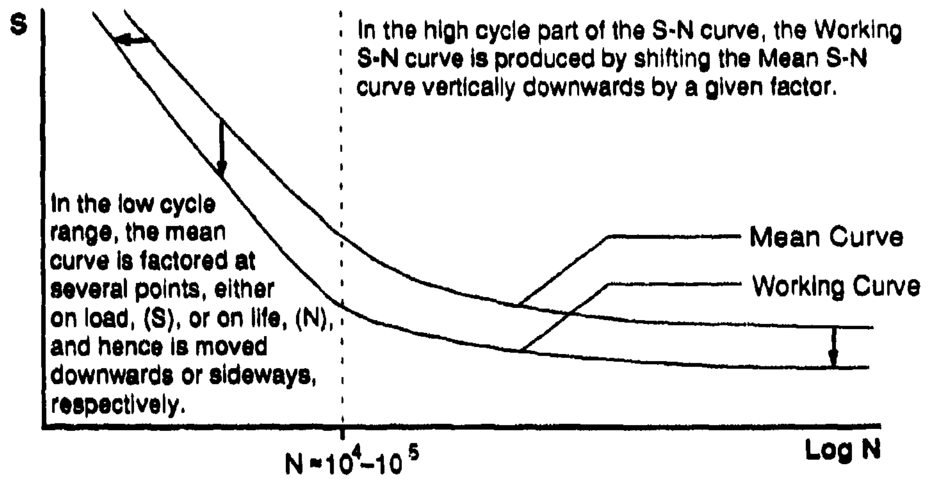

Figure 21: Reduction factors applied to the Mean S-N curve to create the Working S-N curve

Westland Helicopters[26], by way of another example, uses a slightly different approach, but arrives at the same idea of a Working S-N curve. In its component fatigue tests, Westland applies a factor to increase the fatigue loads. Factors vary from 1.40 to 1.85 depending on the material and the number of specimens tested. These factors are applied only to the high frequency loads. The low frequency end of the curve is factored on life.

The difference in the amount of Mean S-N curve reduction amongst the major helicopter manufacturers is shown in Table 4. This table shows the values obtained for the endurance limits of the mean and Working S-N curves in the hypothetical pitch-link fatigue life calculation mentioned in Section 4.1. As can be seen, there is good agreement for the endurance limit on the Mean S-N curve, but quite large disagreements for the Working S-N curve endurance limit. This is the main reason for the large variation in predicted fatigue lives[59].

Table 4
(from Ref. 66)

Values for the endurance limit used in the hypothetical pitch-link exercise

Manufacturer	Mean curve endurance limit (lb)	Working curve endurance limit (lb)
Aerospatiale	2176	1225
Agusta	2100	1674
Bell	2061	1649
Boeing-Vertol	2106	1685
Hughes	2024	1717
Kaman	2101	1615
Sikorsky	2100	1400

* 0.25 cycles is the case in which the specimen is loaded statically to failure. Therefore, the corresponding value of S represents the Ultimate Tensile Strength of the specimen.

4.1.2 Load Spectrum

"Without doubt, the best life predictions will result from tests with actual components and true service loads under realistic environmental conditions", writes Och[56]. This, though, is the exception rather than the rule, as Och admits, especially during the design of a completely new helicopter. Consequently, the load spectrum used in determining the fatigue life of a particular component is derived as indicated in Fig. 19.

The general procedure used in obtaining a load spectrum for a helicopter is as follows:

(i) The manufacturer and/or the prospective customer specify the type of mission, or missions, to be flown by the helicopter. Missions are given general titles such as, for example, Utility or Observation[63]. These missions are specified in terms of "*... the percentage breakdown of all possible flight conditions ...*"[56]. and this percentage breakdown (or mission spectrum) can be defined either coarsely or finely. Examples of very fine definitions of mission spectra are given in Refs 63 and 67. An extract from the mission definitions given in Ref. 63 is shown in Table 5.

Table 5

Examples of Helicopter Missions
(Extract from Ryan et al[63])

CONDITION	Percentage time spent in each condition for the different missions					
	Observation	Utility	Utility/ Assault	Attack	Crane	Transport
GROUND OPERATIONS						
A. Start-up	0.50	0.50	0.50	0.50	0.50	0.50
B. Shut-down	0.50	0.50	0.50	0.50	0.50	0.50
etc						
TAKE-OFF/LANDING						
A. Vertical lift-off and transition to 40 kn	2.66	1.58	0.43	0.22	0.51	0.66
B. Rolling take-off	0.00	0.00	0.43	0.27	1.24	1.53
C. Vertical landing	2.66	1.58	0.43	0.22	0.51	0.66
etc						
ASCENT (> 40 kn)						
A. Steady-state climb	8.36	9.00	10.35	3.66	9.91	9.83
B. Turns	1.45	2.32	4.65	4.34	0.65	0.99
C. Pushovers	0.34	0.34	1.23	1.16	0.00	0.00
FORWARD FLIGHT (>40 kn)						
A. Level flight	38.08	39.02	30.13	33.40	47.58	46.28
B. Turns	8.14	9.91	10.26	11.65	3.02	3.43
C. Control Reversals	1.50	1.27	1.13	0.63	0.10	1.51
etc						
DESCENT (Power on, > 40kn)						
A. Partial-power descent (Steady descent)	2.07	5.26	3.18	2.21	9.10	7.79
B. Dive	3.37	3.50	5.09	3.45	0.05	2.12
etc						
AUTOROTATION (Power off)						
A. Entries (includes power chops)	0.39	0.39	0.39	0.39	0.39	0.39
B. Steady descent	1.63	1.63	1.63	1.63	1.63	1.63
etc						

By its nature, the mission spectrum is an assumption of how a particular helicopter will be used by operators throughout its design life. If the in-service usage of the helicopter differs significantly from that assumed, then the actual fatigue life may be

significantly altered. If the helicopter is a derivative of an existing design, then the manufacturer may have a good idea of the type of usage to expect. On the other hand, for a new helicopter, the "*... definition of the operational spectrum ... normally is made many years in advance of its entry into service, and changes in usage could occur in the interim and have a tremendous impact on the calculated fatigue life ...*", McDermott et al[68]. Also, the definition "*... should provide the broad* [load] *spectra in each role ...* [but] *... the helicopter is a versatile creature and can be used in a wide variety of roles many of which may be devised long after the specification* [of the load spectra] *was prepared and the aircraft is in service ...*"[26].

(ii) The loads acting at each of the flight conditions specified in the mission spectrum must be obtained[53]. There are two ways in which this can be accomplished. The first method is used for a new helicopter design and consists of a combination of analysis and the knowledge of the helicopter designer in knowing the types of relevant loading actions. These loads are determined in general terms (e.g. lift, drag, and c.g. accelerations).

The forces thus determined are then converted to loads which are specific to a particular component. These loads are given in a form which specifies the mean and alternating loads (or equivalents such as maximum and minimum loads) and the loading frequencies.

The second method involves using strain gauges to measure component loads in-flight and is an option if the helicopter is a derivative of an existing design. This may or may not be done, however, depending on how radical are the changes. If the changes are minor, then the designers can be confident that they know the significant loads and so the need to perform flight tests will then be based on cost and time factors.

Flight testing is always used, though, to verify the loads used in the design of a new helicopter. Once a prototype is available for flight testing then a flight loads survey is conducted in which the helicopter is flown through some of the flight conditions specified in the mission spectrum so that corresponding component loads can be measured. The reason that not all of the mission spectrum is covered in the flight loads survey is due to time and cost constraints[85]. The flight test engineers attempt to ensure that at least all the most critical flight conditions are included as well as some of the non-critical ones. Due to physical constraints, not all components of interest can be instrumented with strain gauges, so the manufacturer must determine the relationships between the loads in the instrumented and uninstrumented components. This is usually done by testing of the appropriate component assemblies in static test fixtures.

(iii) In general, the stress ratio, R*, of the loads found in (ii) will not correspond with that of the Working S-N curve. If the spread of flight load R values is not large "*... a representative stress ratio can be assumed for both fatigue test and flight test data, and no further correction is required...*"[69]. If this is not the case, then some type of correction (also known as a reduction) must be applied to the flight loads to make them consistent with the Working S-N curve. Noback[53], indicates that the most common type of mean load correction is the use of the Modified Goodman (or Soderberg) diagram. That is, the damage caused by a cyclic load, S_1 at a mean load, S_{m1} is equivalent to the damage caused by a cyclic load S_2 at a mean load, S_{m2}, where S_{m2} would be the mean load applicable to the Working S-N curve. However, some manufacturers act conservatively and only apply this correction if $S_{m1} > S_{m2}$. Reference 57 shows how Augusta uses the Goodman correction.

(iv) The loads in (ii) are combined with the mission spectrum in (i) to produce the load spectrum for a particular component. The load spectrum is expressed either in percentages of time spent in each flight condition as shown in (i) and/or as the number of occurrences in a given time period (per hour, for example).

* R, the stress ratio for a load cycle, is equal to the minimum stress divided by the maximum stress in the cycle.

4.1.3 Damage Hypothesis

Once the load spectrum and the Working S-N curve have been derived, they are combined by means of a damage hypothesis to obtain a safe-life (i.e. a retirement life) for a component. The hypothesis which is universally used in the helicopter industry is Miner's Cumulative Damage Hypothesis, also known as the Palmgren-Miner Cumulative Damage Hypothesis or, simply, Miner's Rule. The basic method is as follows:*

(i) Select a flight condition from the mission spectrum and hence obtain the corresponding load, S_i

(ii) Determine the number of cycles, N_i, from the Working S-N curve, available at load level S_i before component safe-life is expended.

(iii) Determine the number of occurrences, n_i, of load S_i in the load spectrum for a given time period (say one hour)

(iv) Define the fatigue damage due to load S_i as n_i/N_i per hour

(v) Repeat steps (i) to (iv) for the other loads in the load spectrum

(vi) If there are k loads in the spectrum, then the total fatigue damage, D, induced in the component is:

$$D = \sum_{i=1}^{k} (n_i / N_i) \text{ per hour}$$

(vii) The safe-life of the component is then the reciprocal, i.e. Safe-Life = $1/D$ hours

Miner's Rule, however, is known to have the undesirable trait of sometimes producing unconservative results. It does not, for example, take into account the possible retardation or acceleration effects that large load cycles can have on crack growth[70,71]. This is especially important in helicopters because they are loaded by a mixture of low magnitude, high frequency loads, and large magnitude, low frequency loads. The high frequency loads are mostly below the endurance limit and are therefore ignored by Miner's rule. However, the application of the large magnitude loads can cause the endurance limit to decrease and hence allow the high frequency loads to become damaging (see Section 3.4.3). To offset this, some manufacturers apply a reduction factor to the life calculated in (vii) above for a component if large magnitude, low frequency loads are thought to be important[26,53,57,62]. For example, both Agusta[57] and the American FAA[60] use the following reductions:

- If the calculated fatigue life is less than 3350 hours, the service life is 0.75 times the calculated life.

- If the calculated fatigue life is greater than 3350 hours, the service life is 0.375 times the calculated life plus 1250 hours.

Westland, on the other hand, applies a factor of 0.75 to all component fatigue lives to arrive at the safe-life.

Despite all the shortcomings, the use of Miner's rule has been generally successful because of the vast experience in its application built up since the 1940s along with the use of appropriate empirical correction factors.

4.1.4 Cycle Counting Methods

In Section 4.1.3, it was stated that, to use Miner's Rule, the number of occurrences, n_i, of load S_i must be found. This Section details the three methods which are currently in use.

Assume, for purposes of discussion, that a flight loads survey revealed that during a particular flight condition (call it X), the load time history experienced by a particular

* The method shown assumes that the block counting method is used. However, it is also applicable to other counting schemes in which case a flight condition may have several sets of S_i, N_i, and n_i values.

component was found to be as shown in Fig. 22(a). The load cycles could then be counted by:

(i) Block Counting Method

With the block counting method, the loads generated in the flight condition are taken as a single block and the peak load (X_p) is assumed to occur over the entire time as shown in Fig 22(b). When using the S-N data, X_p is usually taken as $0.5X_p$ or $0.5X_p'$ depending on the manufacturer's design policy[85]. Therefore, there is only one load associated with the flight condition and the number of cycles is n_x.

This is the most conservative of the cycle counting methods. If the lives thus calculated do not yield satisfactory results (i.e. they do not meet minimum requirements), then manufacturers might make use of a less conservative cycle counting scheme.

(ii) Cycle Counting Method I – The "sub-block" approach

The loads in flight condition X are considered in more detail than in the block counting method. The flight condition is divided into several parts as shown in Fig. 22(c) and then the block counting method is applied to each of the resulting "sub-blocks". The actual method of deciding how to create the sub-blocks is not a hard and fast science and consequently each manufacturer has its own preferred schemes.

If the lives produced by this method are still not acceptable, then the number of sub-blocks may be increased, with a consequent decrease in the conservatism of the calculated lives.

(iii) Cycle Counting Method II – The exact cycle counting approach

This is the least conservative of the three methods and is the limit case of the "sub-block" approach in that each individual cycle is counted separately so n = 1 (Fig. 22(d)).

4.2 FATIGUE TESTING

Fatigue testing is an important part of the design, prototype and production phases of a helicopter type. In section 4.1.1, reference was made to the use of fatigue testing in producing Working S-N curves to calculate component fatigue lives during the design process. However, the calculated lives must be verified during the prototype stage[13]. In particular, components which are defined as vital (i.e. those components whose failure would critically affect safety) require full fatigue test substantiation of their calculated lives[64]. Other components, although not in the vital category, may be tested to ensure confidence in their predicted service lives.

There are many different types of fatigue tests from simple component to full-scale airframe tests and many different ways of conducting them: e.g. S-N testing, block loading or spectrum testing. References 24, 23, 28, 12, 61 and 26 contain typical examples. Conventional S-N testing is mainly conducted on simple components whereas block or spectrum loading is applied to complex components. The loads used in the spectrum and block loading are usually based on flight loads obtained from the prototypes and may include a combination of the low and high cycle fatigue loads or may only include the low cycle fatigue loads (e.g. manoeuvre loads, GAG cycle). Kaman Aerospace Corporation recently[23] fatigue tested an SH-2F Seasprite airframe and only applied low cycle fatigue loads because in a previous Seasprite airframe test *"... High Cycle Fatigue was not found to be a limiting factor..."*[23]. However, Westland Helicopters carries out *"... full scale fuselage fatigue tests on the whole fuselage under manoeuvre load conditions. High frequency loading is applied at flight load levels to simulate the fretting condition..."*[26].

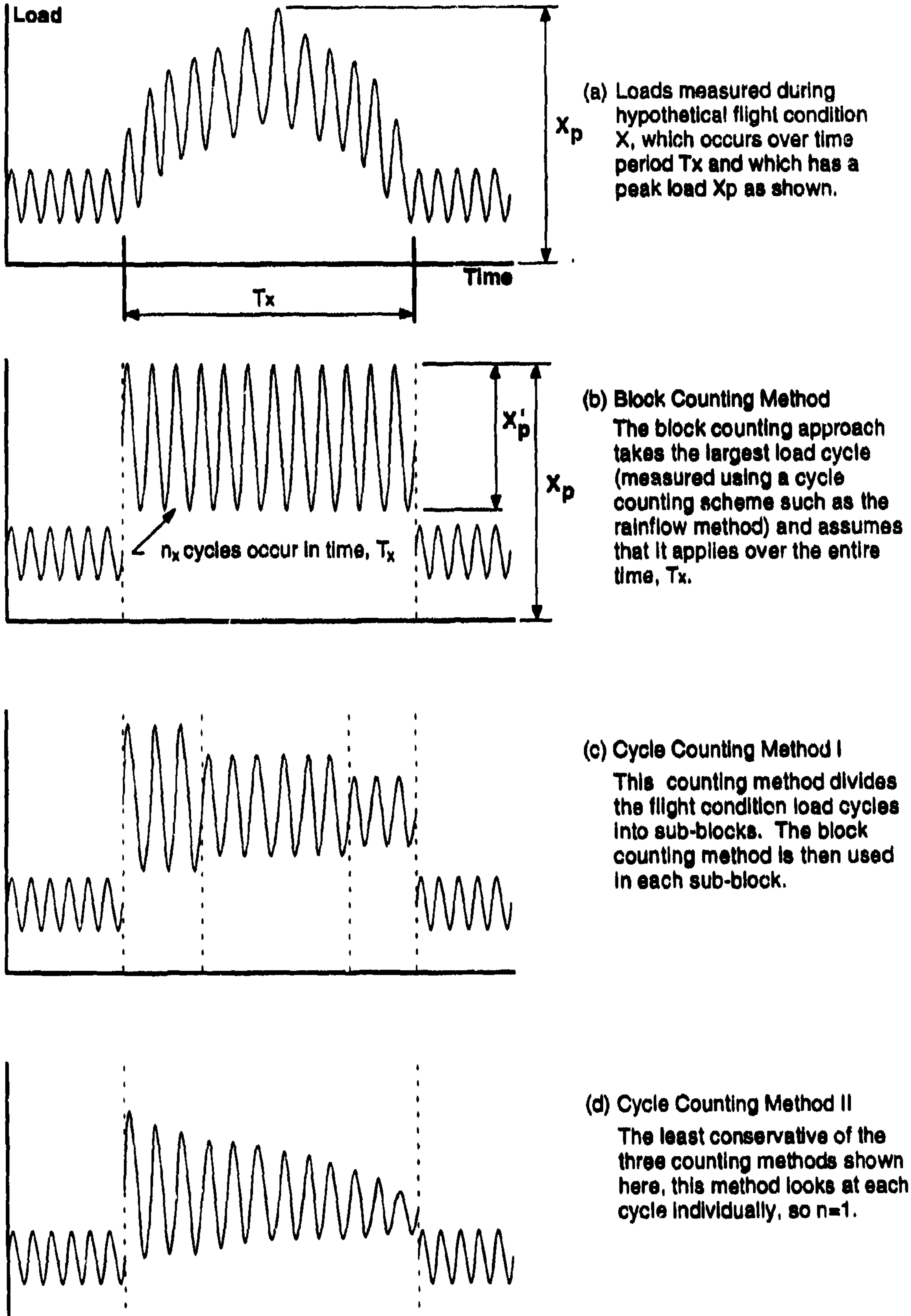

Figure 22: Various cycle counting methods

Full-scale airframe tests are not common. Most fatigue testing is concentrated on sub-assemblies, such as tailbooms and rotor heads, and simple components such as pitch-links. During the development of the Apache attack helicopter, the U.S. Army originally required a full-scale airframe fatigue test. This requirement was, however, dropped after the Army concluded that the airframe test would not be as beneficial as originally thought. *"This conclusion was based on the lack of correlation between helicopter airframe fatigue tests and service experience; and the extensive time and cost required to conduct a meaningful test..."*(Deveaux)[61]. A table showing this lack of correlation is presented by Deveaux in his report and has been reproduced below as Table 6. Even Westland Helicopters, whose practice is to carry out full-scale airframe fatigue tests admits that this type of testing has met with only "... *somewhat moderate success* ..." in determining likely problem areas and repair schemes.

Table 6

Correlation between airframe fatigue tests and service history

Helicopter	Load application method	Total test failures	Non-relevant failures	Relevant failures	Related service failures
AH-56	Airframe loaded	18	6	12	Not fielded
CH-46	Rotor loaded	195	120	75	0
SH-3	Airframe and rotor loaded	24	10	14	14[a]

Notes: [a] Only three design changes were incorporated
Design, fabrication, test, and data reduction required 28 months

Fatigue testing is also used during production of a helicopter type as a quality control measure[26,28,60]. The reason for this is to ensure that the fatigue behaviour of production components does not differ from that of the components used in the verification fatigue tests. The procedure quoted by Hall[26] is regarded as typical[60] and so is presented as an example. Two components are selected from the production line on some predetermined basis, e.g. every 30 production items or once per month. One of these components is then fatigue tested, usually using a simplified version of the original verification fatigue test. Various parameters are determined from the results of this test and compared with "warning limits" and "action limits" which have been statistically determined from previous tests. If all the parameters fall within the warning limits, then the result is deemed satisfactory and the second component is returned to the production line. If any parameter falls outside its action limit, then the component is deemed defective, production is halted and the cause of the problem investigated. If any parameter falls inside its action limit, but outside its warning limit, then the second component is tested. Production is continued only if the results of this test are satisfactory.

The main criteria behind the selection of parts for quality control checks are the complexity of loading and manufacture, and the likely effects of manufacture on the fatigue strength of the part.

5. HELICOPTER HEALTH AND USAGE MONITORING (HUM)

Many and varied techniques have been applied to the problem of monitoring the pattern of usage of a helicopter and the condition of critical components. As indicated by the heading of this Section, the majority of the methods used can be split into two categories: health monitoring (HM) and usage (or fatigue) monitoring (UM). Usage monitoring can be further subdivided into flight loads and flight condition monitoring (Fig 23).

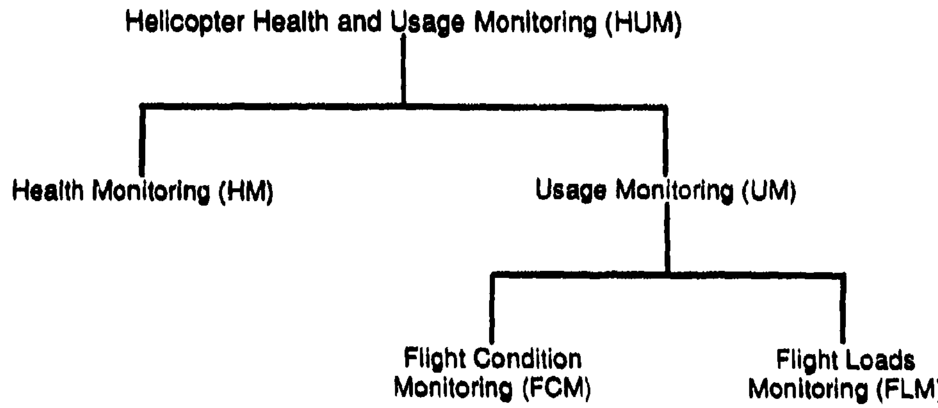

Figure 23: The various categories of helicopter health and usage monitoring

The question arises as to why usage monitoring is necessary or even desirable. The answer is related to the uncertainty that exists over how helicopters are used. As explained in Section 4.1.2, helicopter designers can only make a "best guess" as to the type of missions that the helicopter will fly and then design accordingly. However, the versatility of helicopters means that new uses will be found for the design, especially after it has been in production for a number of years and these uses may go well beyond the originally envisaged design roles.

For example, consider a utility helicopter which is bought by two different operators. The first operator modifies the helicopter for crop dusting and the second uses it exclusively as an executive transport. It is likely that the first operator will find that tail rotor components have to be replaced at hours well below those specified by the manufacturer, while the second operator will have no such troubles. The reason for this lies in the disparity between the missions which these helicopters fly. Tail rotor component fatigue life is adversely affected by time spent in low speed flight or quick 180 degree turns (see Section 3.5). Crop dusting inherently means that a helicopter will spend a large percentage of its flight time in these two conditions: a percentage that will be far greater than that assumed by the manufacturer when the helicopter was designed. On the other hand, the second helicopter will spend its entire life operating at gross weights well below those which it is capable of carrying and will be flown quite gently.

The first operator has an obvious problem: how can the safety of the helicopter be assured, i.e. what are the chances that a component will fail in flight? To solve this dilemma, the operator might voluntarily reduce inspection intervals, but the reduction would have to be quite severe to ensure safety since it would be based only on the operator's experience. This would cause a significant jump in operating expenses. Another solution might be to go back to the manufacturer and request that the safe life of the tail rotor components be recalculated, based on a crop-dusting mission spectrum. This will give the operator a better estimate of the reduced inspection intervals to use, but the analysis would be an expensive exercise. However, what happens during the off-season, when crop dusting is

not required and the operator leases the helicopter for other uses? In this case, do the old or new inspection intervals apply?

The second operator also has a problem, though it is not obvious and is implicitly accepted by every helicopter manufacturer and operator. By serving its life as an executive shuttle, the second helicopter will rarely, if at all, experience any of the more severe manoeuvres and flight conditions which it was designed to handle. Consequently, the safe-life of the components will probably be greater than that indicated by the manufacturer. Thus, by replacing parts at the scheduled intervals the operator is throwing away many still-useful components and therefore incurring operating cost penalties.

The solution for both these operators would be to install a monitoring device on their helicopters. This would establish the severity and type of missions flown or loads experienced which in turn would permit appropriate adjustment of the safe-life accordingly. In the case of the first operator, the monitoring system would show that tail-rotor component safe-lives were being used up quickly. Therefore, the operator could alter how the helicopter is flown to reduce fatigue damage (e.g. by making slower turns), but still being able to complete the crop dusting in a reasonable time and make a profit. For the second operator, the immediate benefit would be the increase in time between component replacements which eventually would translate into lower maintenance and support costs.

Health and usage monitoring can increase helicopter safety and has the potential to decrease operating costs, though the operating cost argument depends on how much the monitoring system will cost to purchase (or develop), install and operate. The magnitude of these costs will depend on the sophistication of the system. Something as simple as a magnetic oil plug for a helicopter transmission casing does not cost much to buy or install in place of a normal plug. The only operational procedure required is to inspect the plug at regular intervals to see if it has picked up any "abnormal" metal particles in the oil. On the other hand, a comprehensive system that monitors dozens of different parameters ranging from component strains to flight conditions would be prohibitively expensive for an operator to purchase and install. Moreover, such systems accumulate vast quantities of data and generally require that the data be down-loaded and sent to a central facility for processing. In this case, the costs of operation will be high since there will need to be personnel assigned to gather and process the data. In addition, there will inevitably be a delay between the time that the data are down-loaded and the time that the processing facility provides an output in a useful form to the operator.

5.1 HEALTH MONITORING

The detection of abnormal behaviour in a helicopter mechanical system forms the basis of health monitoring. As its name suggests, it is somewhat like having a doctor continually checking the health of a person. By watching for symptoms which indicate that trouble is imminent, or at the least, that indicate that cautionary action should be taken, major illnesses and afflictions can be avoided. In a similar manner, health monitoring (HM) systems on helicopters check the health of various components and systems. These HM systems range from the simple and ubiquitous magnetic oil plug to complex ones which analyse gearbox vibrations.

Some health monitoring techniques are used to provide early detection of fracture in some components. In cases where components have a very short life to fracture after a crack has initiated, health monitoring techniques may be inadequate and so the component replacement would be based on the safe-life approach. For most of the components discussed in this report, health monitoring is unsuitable. Rotor blades, though, are a notable exception and a particularly successful application to blade monitoring is the use of pressurised gas inside metallic rotor blades (or the metallic spars of composite blades). A crack growing in the rotor blade skin will quickly become a through-thickness crack, thus allowing the pressurised gas to leak out which causes a loss of pressure inside the blade. The blade internal pressure is monitored by a gauge which is either mounted on the blade

itself (near the hub) where it is regularly checked by service personnel, or in the cockpit, where it immediately alerts the pilot to the problem.

Since HM systems are used mainly in helicopter mechanical systems, they are not within the scope of this report. For further information the reader is referred to papers such as Ref. 73 which contains a good summary of health monitoring techniques.

5.2 USAGE MONITORING

Unlike health monitoring systems, usage monitors look not for abnormal signs, but at the way a helicopter is used. Usage monitoring (UM) examines the usage of a helicopter and then makes an estimate of how much safe-life remains in a component before it must be replaced. UM can be divided into two types as indicated in Fig. 23: flight condition and flight loads monitoring. Usage monitors can be exclusively of one type or the other, or can incorporate elements of both.

UM systems are like HM systems in that they can be simple or complex. Perhaps the simplest usage monitor of all is the flight hours clock, while the most complex UM systems are to be found on flight test or research helicopters.

5.2.1 Flight Condition Monitoring

The aim of flight condition monitoring (FCM) is to identify how severely (or benignly) an individual helicopter is flown by determining the flight conditions which occur during flight. By establishing how the helicopter is used, an estimate can be made of the fatigue life expended in a particular component, provided that some link between flight condition and resultant fatigue damage can be found. This link is generally in the form of an $m \times n$ matrix of transfer functions where m is the number of flight conditions and n is the number of components being monitored. The value of m can, in the extreme, be infinite, but is usually defined to be around 200 - 300 separate and identifiable flight conditions[32]. These conditions are defined in a similar manner to that shown in Table 5. The generation of the transfer function matrix represents a formidable problem. As mentioned in Section 3.2, the current state of helicopter rotor loads prediction techniques is not advanced enough to adequately predict fatigue loads. Therefore, the only practical way at present to obtain the transfer function matrix is flight testing. By strain-gauging numerous components on a flight test helicopter and then having it fly through a series of predetermined manoeuvres, component loads can be measured during different flight conditions. Since helicopter flight testing is an inexact science, the validity of the transfer functions obtained depend a great deal on the ability of the test pilots to fly the "worst-case" situation for each flight condition and so generate the most severe fatigue loads applicable to each component for each flight condition.

This approach is, as Grainger[32] puts it, *"... eminently practical, but is necessarily limited."* The limitations which Grainger refers to are mainly to do with the expense of flight testing. It is an expensive exercise to fly each of the required manoeuvres, especially since it is also desirable to fly each manoeuvre several times to increase the chance that the most severe loads have been obtained. As well, it would be desirable to conduct the flight tests under different environmental conditions, from hot and high, to wet and freezing. However, such tests are not normally done because of the marked increase in the cost of supporting a flight test helicopter away from its factory facilities.

The types of parameters which are measured on an FCM-equipped helicopter can include: control stick positions, altitude, outside air temperature, c.g. position, the presence of external loads, etc. The list can become endless and depends mainly on the type and number of flight conditions which are to be identified. Since FCM relies on the measurement of several helicopter and environmental state parameters, it is sometimes referred to as parametric monitoring.

Once the transfer function matrix is obtained, component loads, and hence fatigue damage, can be estimated. Fatigue damage will normally be estimated by using the same safe-life

techniques as discussed in Section 4. A diagrammatic representation of how a typical FCM system works is shown in Fig. 24.

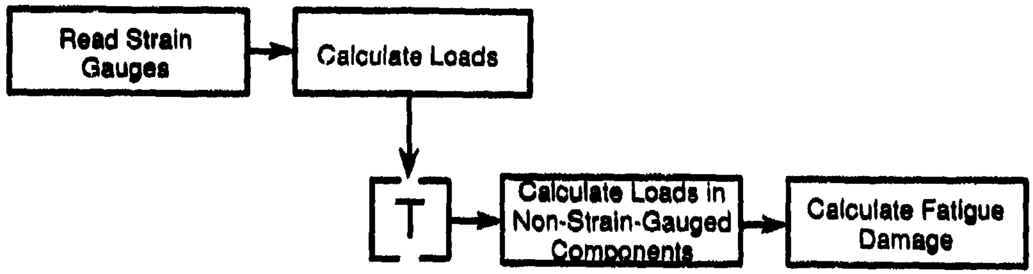

Figure 24: Sequence of events in a flight condition monitoring system. T represents a matrix of transfer functions.

An example of an FCM system which was developed for the U.S. Army during the late 1970s, early 1980s, but not put into operation is the Structural Integrity Recording System (SIRS)[74,75,76]. Another system, this time developed for the U.S. Navy and currently (early 1992) being tested on an AH-1W Cobra is the Structural Data Recording Set (SDRS)[77]. This system is made by Systron Donner, a division of THORN EMI, and is scheduled to enter full-scale production in 1993. Current plans call for the SDRS system to be installed on all fixed and rotary-wing aircraft in the U.S. Navy inventory.

5.2.2 Flight Loads Monitoring

Flight loads monitoring (FLM) is similar to FCM in that they both attempt to quantify in-flight fatigue damage. Unlike FCM, however, FLM does not rely on indirect methods to determine loads, but on the output from strain gauges which are mounted on several key components. Transfer functions are still used, though, to estimate the loads on components that are either impractical, inaccessible, or not necessary to strain gauge. A diagrammatic representation of FLM is shown in Fig. 25.

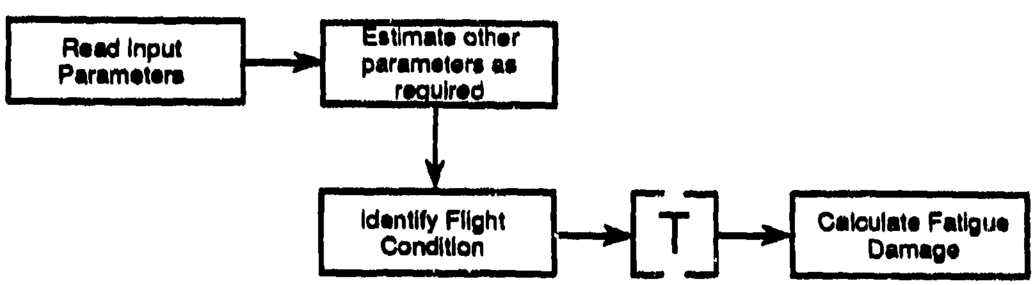

Figure 25: Sequence of events in a flight loads monitoring system. T represents a matrix of transfer functions.

The components which are strain gauged are mainly those in the rotating system since there are the ones most prone to fatigue failures. This immediately means that the strain gauge signals must be transferred from the rotating system to the fixed system (i.e. the airframe) since that is where the recording and analysis system would be located. This has been achieved either through the use of slip rings[39,78] or radio telemetry. As Holford[39]

points out, these methods may be acceptable on flight test and research helicopters, but they are likely to cause maintenance problems on in-service aircraft.

Once the strains have been acquired, they are converted to loads. These loads are then used to derive loads in non-strain-gauged components via the use of transfer functions. After this, the fatigue damage is determined in the same way as is done in an FCM system.

FLM has been restricted to applications involving testing and research because of the problems associated with transferring strain gauge signals across from the rotating system to the airframe. However, there are other approaches to FLM which use indirect measuring techniques similar to those used for FCM. Rotating component loads can be deduced from airframe load measurements. By mounting strain gauges only on airframe components, the need for slip rings is eliminated. The penalty incurred is that now yet another set of transfer functions must be found which link loads in the fixed system to loads in the rotating system. Such an approach has been tried by the Canadians[72] with some success and is currently being investigated by Kaman Aerospace[79,83] under contract to the U.S. Army.

Another method being investigated by the U.S. Army is an FLM system known as SIMS (Smart Integrated Microsensor System)[80]. SIMS essentially consists of a self-contained strain-gauge and micro-processor unit which will be permanently attached to any desired component. The strain gauge signals are fed directly to the microprocessor which is pre-programmed with a cycle-counting scheme and a safe-life analysis to determine fatigue damage. The unit will have some type of display which will indicate the percentage of life expended. This is different to most FLM systems in that the input data are processed on-board and the only output is a single number - the percentage of safe-life expended.

5.2.3 Flight Condition versus Flight Loads Monitoring

The following list summarises some of the advantages and disadvantages of the two systems.

- FCM is, in a sense, compatible with the way that a helicopter is designed. As shown in Section 4, a helicopter design is based on a mission which is defined in terms of the percentage time spent in various flight conditions and not in terms of component loads.
- The amount of conservatism in FCM depends critically upon the transfer functions which convert flight condition to component loads. If the flight tests do not achieve the most damaging conditions for each flight condition, then the transfer functions will yield low component loads under some circumstances. The chances of this happening depend on the experience of the test pilots and flight test engineers. The main way of counteracting this is by using safety factors in the safe-life calculations.
- By using strain readings, FLM systems eliminate the uncertainty of the FCM transfer functions. Strain gauges can be mounted on components such that they read the nominal component strain and then an appropriate stress concentration factor can be applied to give peak strains. The amount of uncertainty in detecting peak loads is less as is the possibility of error in the transfer functions needed to determine loads in non-strain-gauged components.
- A high-fidelity flight regime recognition algorithm is essential for FCM systems. The algorithm must be capable of recognising all the required flight conditions. However, the recognition of low speed manoeuvres in particular has always been difficult. This problem is yet to be adequately addressed.
- An FCM system should be easier and less expensive to install than an equivalent FLM system. This comes about because some of the required inputs for an FCM system are normally already available in a usable form. In modern helicopters, with digital automatic flight control systems (AFCS), most of the required input parameters are read by the AFCS so an FCM system can "tap in" to the AFCS wiring and read the existing signals. FLM systems require the installation of strain gauges and these have to be installed with care. If the FLM system has strain gauges mounted on rotating

components, extra care needs to be taken in mounting the gauges. In addition some means, such as slip rings, must be provided to bring the strain gauge signals down to the airframe (i.e. the fixed system).

- Strain gauge signals can become corrupted by electrical noise as they are transferred across slip rings so this requires the addition of filters or error correction software to reduce the amount of bad data.
- Routine rotor head maintenance of helicopters equipped with an FLM system will become more complicated because of the presence of strain gauges and slip rings.
- FCM systems can provide a history of how the helicopter was used in a way that is easily understood, i.e. percentage times in various flight conditions and the resulting component loads. Such knowledge is useful in setting up mission spectra for new and derivative helicopter designs, but it is especially useful for helicopter operators. By being able to correlate actual usage with fatigue damage, operators have the chance to alter their usage to minimise damage and hence operating costs.
- FLM systems can produce a spectrum of the loads experienced by a particular component. This information can be useful to helicopter designers.
- Some FLM systems that are currently being researched propose doing away with strain gauges in the rotating system and have them only in the fixed system. This reduces the expense of an FLM system, but introduces another level of calculation (and uncertainty) into the process by having to determine rotating system loads from fixed system loads

5.2.4 Which System?

There is no clear-cut answer as to whether flight condition or flight loads monitoring is better, or whether a hybrid system, combining aspects of both FCM and FLM should be used. The decision on what to use depends on the reasons behind the proposed monitoring program. For example:

- What are the aims of the monitoring program? That is, is the monitoring program meant to give insight into how a helicopter, or fleet of helicopters, is flown? Perhaps it is meant to validate calculated fatigue lives in a new design, or maybe to modify existing component replacement times?
- Is the system going to be installed on a number of helicopters or only on a research or test vehicle?
- Is the direct measurement of flight loads important or are indirect methods acceptable?
- Do the measured data have to be stored, i.e. is a recorded history (either loads or flight condition) necessary?
- Is a correlation of flight condition with component loads required?
- Will the monitoring system process all the input data onboard or will the data have to be sent somewhere for processing?

The answers to these questions will provide the basis for choosing the type of system that should be used and its level of sophistication.

If only one component on a helicopter is of concern then it would be an overkill to install a complete FCM and FLM system. A simple FLM or FCM system would suffice. For instance, when engine transmission shafts on U.S. Army CH-47D Chinooks were experiencing fatigue failures the problem was thought to be excessive variation in engine torque; so a simple system, consisting of a one-channel recorder, was installed to monitor and record engine torque. The results of this investigation[81] showed that engine torque did indeed vary and by a substantial amount. On the other hand, a prototype or research helicopter will probably need to have a combined FCM and FLM system.

For a monitoring program designed to more effectively use a fleet of helicopters, a system incorporating FCM would be better for three main reasons:

(i) Current FLM systems would be more troublesome in terms of maintenance and operation as explained previously. If systems such as the one being studied by Kaman Aerospace[79] are successfully developed, then these penalties will be substantially reduced.

(ii) The expense involved with installing and maintaining an FCM system would, in most cases, be less than an FLM system. This becomes important when the system is to be deployed on many helicopters.

(iii) The results obtained from an FCM system would be more useful to the fleet operator since they can be directly related to how the helicopters are flown.

5.2.5 What to monitor?

(a) An FCM system would need, as a minimum, the following inputs:

- Airspeed: slow speed and high speed flight are particularly damaging flight regimes.
- Weight: high gross weight flights will incur more fatigue damage.
- Pressure Altitude and Outside Air Temperature: gross weight and true airspeed cannot be determined unless these two parameters are known.
- Centre of gravity position: adverse c.g. positions accentuate fatigue damage.
- Collective and cyclic stick positions: required to determine the type of manoeuvre.
- Vertical acceleration: required to determine the type of manoeuvre.
- Main rotor RPM: measurement of this will indicate when a GAG cycle occurs.
- Yaw rate: high yaw rates and yawed flight are damaging to tail rotor components.

Depending on how fine the separation of recognisable flight conditions is, other parameters such as pitch and roll rate, lateral and longitudinal acceleration, tail rotor pedal position, etc, might be required.

If the FCM system is to also monitor the usage of mechanical components, then other parameters, such as engine torque, would need to be added to the minimum list shown above.

(b) An FLM system only monitors strains so the only input requirements are strain gauge signals. It is not possible to give a definitive list of components to monitor since this depends on the exact nature of the load paths in the structure. However, some generalisations may be made about where to mount strain gauges:

- Rotor system components will have much shorter fatigue design lives than airframe components. Table 7 is a list of the fatigue-critical components on an AH-64A Apache and, of the 21 items listed, 20 are from the rotor system. Hence, if fatigue problems occur in a helicopter, they are more likely to manifest themselves in the rotor system. The actual components to strain-gauge in the rotor system will depend upon their relative fatigue lives, their importance to flight safety and the ease with which they can be accessed in order to apply the strain gauges.
- Lift frames carry the entire weight of the airframe and payload during flight so they may be candidates for strain-gauging.
- Many helicopters are designed such that their main rotor gearbox (MRGB) housings are part of the airframe structure. This means that the housings are a link in the load path from the rotors to the airframe. The points where the airframe attaches to the MRGB housing are, therefore, possible sites for fatigue cracking.

- Any frame which carries concentrated loads (e.g. landing gear loads, sling loads and winch loads) may experience fatigue cracking or induce cracking in nearby structure.
- A tailboom can be susceptible to fatigue cracking, especially around the area where it joins onto the fuselage. The most susceptible types of tailbooms are those which are very long and slender, have heavy masses located at their ends and have no tail wheel to support them during landing.

Table 7
(from Ref. 82)

List of fatigue critical components on the AH-64A Apache

Item		Location
1.	Main rotor blade root end doubler	Rotor System
2.	Main rotor blade trailing edge	Rotor System
3.	Main rotor blade attachment pins	Rotor System
4.	Lead-Lag link	Rotor System
5.	Lead-Lag link damper lugs	Rotor System
6.	Lead-Lag damper trunnion	Rotor system
7.	Lead-Lag damper body	Rotor system
8.	Main rotor pitch link	Rotor System
9.	Main rotor driveshaft (torsion)	Rotor System
10.	Main rotor longitudinal bell-crank lug	Rotor System
11.	Main rotor lower swashplate bearing retainer	Rotor System
12.	Main rotor collective bell-crank	Rotor System
13.	Main rotor lateral bell-crank	Rotor System
14.	Pitch housing lead-lag lug	Rotor System
15.	Pitch housing trailing edge damper lug	Rotor System
16.	Pitch horn	Rotor System
17.	Tail rotor fork	Rotor System
18.	Tail rotor output driveshaft	Rotor System
19.	Tail rotor quill shaft	Rotor System
20.	Tail rotor controls bracket	Rotor System
21.	Tailboom (bending)	Airframe

5.2.6 The Need for Usage Monitoring

Helicopter operators, both civil and military, are ever conscious of the cost of running their fleets. Thus, the decision to use or not use usage monitors would be driven by economic factors*. It is not possible to make a blanket statement about the necessity or otherwise of usage monitoring for helicopter operators, but it is possible to make some general remarks. These remarks are all based on the first question that an operator must ask when contemplating usage monitoring: *"Why are components being replaced?"* The possible answers to this question are:

(i) Component replacements are occurring for reasons other than fatigue (such as corrosion for example).

(ii) Components are replaced because they reach their safe-life retirement lives.

(iii) Components are being replaced before their safe-life retirement lives because of unexpected fatigue cracking.

If the operator is changing components for reasons other than fatigue (case (i)), then usage monitoring would be of little or no use. Usage monitors do not identify problems, such as corrosion or erosion, that weaken components.

* In addition, civil operators face various legal requirements placed on them by airworthiness authorities, but these are not relevant to this report and so will not be discussed.

If the operator finds that cases (ii) or (iii), or a mixture of cases (i), (ii), and (iii) apply, then usage monitoring might be of benefit. Case (ii) implies that the helicopter operator is using its helicopters less severely than the manufacturer anticipated would be the case. In this case, usage monitoring may enable component retirement lives to be extended, thus reducing operating costs, but there is an important caveat attached. That is, if usage monitoring indicates that components can be operated past the manufacturer's stated retirement lives, then there must be agreement from the manufacturer to do so. This is perhaps an obvious point, but it must be stressed that usage monitoring technology, no matter how sophisticated, does not give helicopter operators the authority to bypass the manufacturer and decide on component retirement lives by themselves.

There are two possible ways in which component safe-lives could be extended. The first way is to extend the life of a particular component by the same amount across the fleet. In order to illustrate this, assume that a particular operator has installed usage monitors on all its helicopters and that the operator is examining the impact of usage on the main rotor shafts which have a manufacturer specified retirement life of 2000 hours. Further, assume that the results of the usage monitoring show that the rate of main rotor shaft fatigue damage accumulation across the fleet is, at worst, 75% of that assumed by the manufacturer. This indicates to the operator that there is room to increase the specified retirement life without compromising safety. Hence, if the manufacturer agrees, the retirement lives of all main rotor shafts in the fleet could be raised by a conservative amount, say 15 % (i.e. 2300 hours). This method of life extension requires a large data base of information on the operator's fleet usage to ensure that any life extensions are, in fact, conservative.

The second way of extending component lives is to base retirement on an individual usage basis. That is, to use the previous example, instead of applying a new retirement life of 2300 hours to all the main rotor shafts in the fleet, each helicopter's usage monitor would determine when the shaft on that particular helicopter had expended its safe life and needed replacing. This could mean that one shaft might be replaced at 2600 hours, another at 2700 hours, yet another at 2750 hours, etc. This life extension philosophy of replacing components based on monitored usage represents a radical departure from the current practice of replacment based on the number of hours flown. However, this philosophy is also one that has been widely practised for many years on helicopter engine components.

In case (iii), usage monitoring could be used to determine why the unexpected fatigue cracking is occurring. It is possible that it is happening because the operator is flying a more severe mission than that which the helicopter manufacturer assumed would be the case. If this is so, then a usage monitor would be able to help by allowing the operator to track down the flight conditions which are causing the accelerated fatigue damage. This would increase flight safety and also allow the operator the chance to change flight profiles so that the damaging flight conditions are avoided as much as possible.

However, the reasons for component replacements only give the operator a tentative idea about the usefulness of usage monitoring. The actual need for it would be based on a cost and benefits analysis and whether the manufacturer was willing to cooperate. Such an analysis would need to take into account the value of the components that were being replaced for safe-life reasons (cases (ii) and (iii)) versus the value of components being replaced for other reasons (case (i)), the potential reductions in operating costs, the safety aspects of having a usage monitoring system, and the actual cost of installing and operating the system.

Usage monitoring is a field that will grow in the next few years as operators and manufacturers strive to reduce helicopter operating costs without compromising safety.

6. CONCLUDING REMARKS

This report presents a review of some of the structural integrity issues relevant to helicopter fatigue. The important points to emerge from this review are:

- Helicopter rotor systems accumulate vast numbers of load cycles, typically more than 10^4 cycles per flight hour unlike fixed-wing aircraft which generally experience only a handful of load cycles per flight hour.

- The amount of time that a helicopter is in a particular flight condition is as important as the flight condition itself. Accumulation of fatigue damage is proportional to the time spent in the flight condition.

- The critical fatigue load cases of helicopters differ from those for fixed-wing aircraft because helicopters use rotating wings (rotor blades) to generate lift. The rotating action of the rotor blades requires that the amount of lift that they produce vary in a cyclical manner which is quite unlike the wings on a fixed-wing aircraft. These cyclic variations in the lift along with the aerodynamic interference between the rotor blades and the airframe, and the ability of the helicopter to hover, fly backwards or sideways, and take off or land vertically, all conspire to complicate the fatigue loading environment for helicopters.

- The safe-life approach for fatigue design is the method of choice in the helicopter industry. The high number of applied load cycles and the single load path characteristics typical of helicopter dynamic components generally preclude the use of the safety-by-inspection methods of the durability and damage tolerance approach.

- Helicopter usage monitoring has hitherto been used for verifying mission spectrum assumptions or for research. However, it has the potential to improve in-flight safety and reduce operational and support costs for helicopter operators.

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GLOSSARY

Advancing Blade

The tangential velocity of a rotor blade can be divided into two components: one parallel, and the other perpendicular, to the motion of the helicopter. If the parallel component is acting in the same direction as the helicopter c.g. velocity then the blade is said to be an advancing blade. If the opposite is true, namely, that the parallel component is acting opposite to the helicopter c.g. velocity, then the blade is called a retreating blade. By definition, a helicopter hovering in still air has neither an advancing nor a retreating blade.

Anti-torque Force

An engine which applies a torque to an object (e.g., an engine driving a helicopter rotor) will tend to rotate in the opposite direction to that torque. Since in a helicopter the engine is attached to the airframe, then the airframe will also tend to turn in the opposite direction to the applied torque. To stop this, an anti-torque force is required. There are many ways to achieve this, the most common being tail rotors or twin main rotors.

Articulated Rotor

Rotors in which the blades are attached to the hub by hinges or bearings, to allow flapping, pitching, and lead/lag motions, are known as articulated or fully articulated rotors.

Autorotation

If a helicopter loses engine power, it will immediately begin falling as the rotor thrust drops off. When this happens, the pilot will alter the rotor blade pitch angles so that they can begin generating lift from the airflow which is now coming up from underneath the rotor. In effect, the rotor becomes a windmill and extracts energy from the airflow to generate sufficient lift to slow the rate of descent of the helicopter to a safe speed.

Blade Passing Frequency

A helicopter with b blades which are rotating at a frequency Ω has a blade passing frequency of $b\Omega$. In simple terms, it indicates how often a point under the rotor sees a blade pass overhead.

Blade Radius

The radius of a rotor blade is measured from the centre of rotation to the tip of the blade.

Blade Sailing

Blade sailing is a term used to describe the behaviour that rotor blades can exhibit, during rotor shut-down, after landing. Rotor blades can be thought of as flexible beams which obtain their required stiffness during flight through the action of centrifugal forces. However, after landing, as the rotor blades slow down, the pilot loses effective control over them since their stiffness decreases dramatically. In this situation, a sudden gust of wind can cause the blade to make large up and down flapping motions. This can cause the blades to hit the ground and shatter, chop through tailbooms or strike any hapless individual unlucky enough to be inside the rotor radius. Blade sailing is more likely if the rotor blade construction is very flexible and if the rotor head is articulated.

Chordwise	Refers to a direction or force which is in the plane of motion of a particular rotor blade. Bending moments which cause bending in the chordwise direction are referred to as chordwise (or chord) bending moments. *See also: Flapwise*
Coaxial rotors	A helicopter with coaxial rotors has two main rotors mounted one on top of the other. They are made to rotate in opposite directions, thus cancelling the torque effects of the engine on the airframe. *See also: Anti-torque Force*
Collective	The collective refers to the components in the rotor system that control the amount of constant pitch angle applied to the rotor blades. When the collective stick is moved, the pitch of all the blades changes by the same (i.e. by a collective) amount. The collective controls the amount of thrust produced by the rotor. *See also: Swash Plate*
Collective Stick	The cockpit control stick that controls the amount of collective pitch.
Cyclic	The term "cyclic" refers to the components in the rotor system that control the range of pitch angles through which a blade moves as it rotates through 360° about its rotor shaft. A helicopter without cyclic controls would roll over because its advancing blades would generate more lift than its retreating blades. With cyclic controls, the blade angles can be varied so that the lift produced by all the blades is balanced. The cyclic controls also determine the direction of motion of the helicopter by tilting the rotor thrust vector in the appropriate direction. *See also: Advancing Blade, Swash Plate*
Cyclic Control Stick	The cockpit control stick that controls the amount of cyclic pitch. It is analogous in its function to the control stick in a fixed-wing aircraft.
Drag Damper	Another term for Lead/Lag Damper
Disc Loading	The ratio obtained by dividing the weight of a helicopter by the rotor disc area.
Drag Hinge	Another term for Lead/Lag Hinge
Dragwise	Another term for Chordwise
Edgewise	Another term for Chordwise
Flapping	The blades on a rotating helicopter rotor do not remain in a fixed plane perpendicular to the rotor shaft. Instead, the blades move up and down as they rotate in order to keep the lift, inertial and gravitational forces in equilibrium. This motion is called flapping.
Flapwise	Refers to a direction or force which is perpendicular to the plane of motion of a particular rotor blade. Bending moments which cause bending in the Flapwise direction are referred to as flapwise or flap bending moments. *See also: Chordwise*
Flatwise	Another term for Flapwise

Ground Resonance	A condition in which the motion of the centre of gravity (c.g) of a rotor couples with the sideways or fore and aft motion of the rotor shaft. If insufficient damping is present, the helicopter will rock from side to side, out of phase with the movement of the rotor c.g. The condition mainly occurs on the ground since then the helicopter is resting on its undercarriage, which provides only light damping against rocking motions. If ground resonance begins, it can cause the disintegration of a helicopter within seconds. *See also: Lead/Lag Damper*
Ground-Air-Ground Cycle	The Ground-Air-Ground (GAG) cycle consists of the large variation in load which occurs once per flight. The minimum load for a GAG load cycle is either zero or the largest negative load seen by the rotor system on the ground or during the flight. The maximum load is the highest positive load experienced by the rotor system during the flight, be it a flight load or a landing load.
Hinged Rotor	Another term for Articulated Rotor
Hingeless Rotor	Another term for Rigid Rotor
Hub	The rotor hub is at the centre of the rotor system. It comprises the blade control system (linkages, swash plate etc) and it transmits engine torque to rotate the blades as well as reacting the resulting blade loads.
Jump Take-off	If a helicopter takes off vertically from its landing pad without ground taxiing first, it is said to be using a jump take-off.
Lagging	When a helicopter is hovering in still air, the blades that make up its rotor, when viewed from above, are equally spaced. However, when the helicopter begins to move horizontally in any direction, the blades are no longer equally spaced. When compared to their positions in hover, some of the blades will be lagging behind where they were in the hover while others will be ahead (i.e. leading). This leading or lagging motion occurs in a chordwise sense.
Lag Hinge	Another term for Lead/Lag Hinge
Leading	See Lagging
Lead/Lag Damper	This is the damper which is installed in rotor heads to prevent ground resonance from occurring. It achieves this by damping the lead/lag motion of the rotor blades.
Lead/Lag Hinge	The hinge mounted at the root of a rotor blade which allows the blade to lead or lag.
Lift Frames	These are the frames in a helicopter which provide the load path for the main rotor thrust to lift the weight of the airframe and payload.

Nap-of-the-Earth Flight	Helicopters are more vulnerable to anti-aircraft defences than fixed-wing aircraft. To combat this, military helicopter pilots fly in what is termed NOE (Nap-of-the-Earth) flight. This consists of the helicopter flying at high speed at or below tree-top height and following the contours of the ground.
Rigid Rotor	A rotor in which the blades are rigidly connected to the hub and the hub is rigidly connected to the rotor shaft is known as a rigid rotor. That is, unlike an articulated rotor, the blade flapping, pitching, and lead/lag motions are not accommodated by hinges or bearings. Instead, the blade motions are catered for by beams which have stiffnessess tailored to allow the various motions.
Root Cut-out	The rotor blade does not physically start from the centre of rotation. There are several reasons for this: the purely physical problem of several rotor blades attempting to occupy the same point in space is one, while another is the desire to reduce the size of the stalled flow region near the hub. Hence, the term "root cut-out" refers to the distance from the centre of rotation to where the blade physically begins.
Rotor Engagement	When a helicopter starts its engine, the rotor is disconnected from the drivetrain until the engine reaches its operating condition. At that time, the drivetrain is connected to the rotor to begin driving it. This is called rotor engagement.
Retreating Blade	See Advancing Blade
Rotor Disc Area	If R is the blade radius then the rotor disc area is defined as πR^2.
Rotor Shaft	The shaft which transmits engine torque from the engine to the hub.
Run-on Landing	A run-on landing occurs when a helicopter lands with some forward velocity.
See-saw Rotor	Another term for Teetering Rotor
Semi-articulated Rotor	A semi-articulated rotor has some of the characteristics of an articulated rotor and some of the characteristics of a rigid rotor. That is, pitching and lead/lag motions may be accommodated by hinges while flapping may be accommodated by the deflection of a beam.
Sideflare	A sideflare is a the same as a normal landing flare except that it occurs in a sideways direction.
Swash Plate	The swash plate is a key component in the rotor system. It converts control system commands from a non-rotating system (the pilot controls) to a rotating system (the rotor).
Tail Rotor	A tail rotor is a small rotor mounted at the rear of a helicopter which runs at five to ten times the speed of the main rotor. Its main function is to provide the required anti-torque force by generating a counter-acting side thrust. *See also: Anti-torque Force*

Tandem Rotors

A tandem rotor helicopter has two main rotors turning in opposite directions in the same way as a helicopter with coaxial rotors and for the same reason. The difference is that the rotors are not mounted coaxially, but, instead, one is situated at the front of the helicopter and the other at the rear (i.e. a tandem arrangement).

Teetering Rotor

A rotor in which the blades are rigidly connected to the hub, but the hub is free to tilt with respect to the rotor shaft, is known as a teetering rotor. It has been widely used on Bell helicopters.

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Rotors		

16. ABSTRACT

This report is a review of traditional practice with respect to helicopter structural integrity. Aspects covered are: significant fatigue loads for the airframe and rotor system, the methods used in the fatigue design of current and previous generation rotorcraft, fatigue test requirements, and health and usage monitoring methods.

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16. ABSTRACT (CONT).

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